CONTRACT NO. NAS5-478

FINAL ENGINEERING REPORT

TIROS II

METEOROLOGICAL SATELLITE SYSTEM

VOLUME II

Prepared For The

NATIONAL AERONAUTICS
AND SPACE ADMINISTRATION

By The

ASTRO-ELECTRONICS DIVISION
DEFENSE ELECTRONIC PRODUCTS
RADIO CORPORATION OF AMERICA
PRINCETON, NEW JERSEY

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WASHINGTON, D. C.

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PRINCETON, N. J.

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PREFACE

This is the Final Engineering Report for the TIROS II Meteorological Satellite System, which was developed by Astro-Electronics Division of the Radio Corporation of America for the National Aeronautics and Space Administration. The Report is issued in accordance with the requirements of NASA Contract No. NAS5-478.

This Report provides technical descriptions of the design improvements and the additions made to the TIROS satellites and ground stations in preparing them for use in the TIROS II Meteorological Program; describes the various system and subsystem tests, and the environmental tests; and describes the prelaunch and launch phase activities at the TIROS ground stations. The post-launch operations and an evaluation of the performance of the TIROS II satellite will be presented in a TIROS II Post-Launch Evaluation Report.

The background of the TIROS project is discussed in the "Final Comprehensive Technical Report, TIROS I Meteorological Satellite System." The "Post-Launch Evaluation Report" for the TIROS I Meteorological Satellite System contains detailed discussions of the evaluation results. These results led to the design improvements which were subsequently incorporated in the TIROS II satellites and ground stations.

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

PART 3. TESTS

TABLE OF CONTENTS (Cont'd)

PART 4. FIELD OPERATIONS

TABLE OF CONTENTS (Cont'd)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Cont'd)

LIST OF ILLUSTRATIONS (Cont'd)

PART 3. TESTS

SECTION I. GROUND STATION TESTS

Ground station tests were limited primarily to the components that were either modified, or specially designed or procured for the TIROS II system. The components were initially tested in the TIROS backup station at RCA-AED. After the components were installed in the vans that were to comprise the PMR station, or integrated into the existing equipment at Fort Monmouth, tests were conducted in the presence of NASA representatives to prove the operability of the ground stations.

The acceptance tests on the PMR equipment were conducted at the RCA-AED facility near Princeton, New Jersey. These tests were conducted under conditions that simulated the environment which would be experienced when the equipment was installed at PMR. The prototype TIROS II satellite was set up within the RCA-AED building, and an RF link was established between the satellite and the equipment vans (located in a courtyard outside the building).

The vans housed all of the ground station components with the exception of the video monitor rack*, the sun-angle computer, and the control-tone demodulator. Those components were installed in the AED building. A coaxial cable and a twisted pair were used to interconnect the vans and the components in the building. This was done to simulate the microwave link which would be used to interconnect the two sets of equipment when they were installed at PMR.

After the test set up was completed, the satellite was programmed for direct and remote picture taking sequences. The TV picture data was transmitted from the satellite to the vans via the RF link and then relayed to the video monitor rack via the coaxial cable. Source data (Direct 1, Playback 1, etc.) was sent through the twisted pair to the control-tone demodulator, which converted the source data into relay closures for application to the sun-angle computer.

The operation of the attitude pulse selector was demonstrated by using a beacon transmitter whose subcarrier was deviated by an attitude pulse simulator. A local noise generator was used to simulate the spurious pulses that could be expected during the earth period of an

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* The video monitor rack consisted of the monitor control, the tape and computer control the monitor oscilloscope, the display and video amplifier, the sawtooth and deflection unit, the horizontal sync separator, the TV-FM demodulator, and the related power supplies. The components were housed in equipment racks 4 and 5.

actual satellite scan. In addition, the horizon sensor in the satellite was activated by a heat source to demonstrate the compatibility of that system with the satellite's beacon transmitters.

A pre-recorded, IR composite-signal, test tape was used to demonstrate the operation of the quick-look demodulator, the IR buffer, and digital time-measuring device. The actual test consisted of two phases. During the initial phase the IR composite signal was played into the quick-look demodulator and the output of the demodulator was recorded on the ground-station tape recorder (No. 2) at a 60 inch-per-second rate. The second phase of the test consisted of playing back the data recorded on the ground-station tape recorder at 3-3/4 inch-per-second rate. This played-back data was applied through the IR buffer to the DTMD and punched-out on paper tape by the DTMD's tape punch.

The equipment successfully passed all phases of the acceptance tests.

Testing of the Fort Monmouth CDA station was performed at the Fort Monmouth installation. Therefore, no "simulated-environment" test set-ups were required. The acceptance test procedures were identical to the PMR test procedures. The Fort Monmouth CDA station was successful in passing all phases of the acceptance test.

SECTION II. SATELLITE COMPONENT TESTS

Only those components that were not government furnished (that is, residual from TIROS I) were subjected to component level tests. The items of government furnished equipment, having passed all the test requirements during TIROS I, were inspected to make sure that they had not been damaged during storage or previous use. Off-the-shelf component items were purchased only after being recommended by the Central Standard Engineering Department of RCA. Upon receipt of each group of satellite parts, a 100-percent quality inspection and test was conducted by the Purchase Material Inspection (PMI) activity to ensure that the parts met specifications.

Component level environmental testing of the new components were conducted during the design and development phase of the TIROS II. These tests are described in Part 2, Development and Design.

SECTION III. SATELLITE SUBSYSTEM TESTS

A. SPECIFIC-PERFORMANCE-EVALUATION TESTS

1. Introduction

The specific-performance-evaluation tests provided complete data for evaluating the operation of the satellite's subsystems. Because of the nature of these tests they were performed only on the separate subassemblies and were not repeated on the integrated TIROS II satellites. This special group of tests included the despin tests, camera response tests, solar-cell output tests, and thermal tests.

2. Despin Tests

The TIROS II despin tests were conducted on both a dummy satellite and on flight model satellite F-2. The basic concept of the despin, or Yo-Yo, mechanism had been proven by TIROS I testing and by the successful despin of the TIROS I satellite. TIROS II despin testing was aimed at determining the change in despin weight which would be required to compensate for the difference between the moments of inertia of the TIROS I and TIROS II satellites, as well as at establishing a more accurate means of predicting "in orbit" despin ratios. The philosophy of despin testing is described in Volume III of the TIROS I Final Report (Reference 1).

The despin tests were performed using the same test set up as had been used during the TIROS I tests. The results of the valid TIROS II despin tests are tabulated here:

TABLE 7. RESULTS OF DESPIN TESTS

Test Number	Satellite	Moment of Inertia	Initial Speed (ω_i)	Final Speed (ω_f)	ω_f/ω_i
2	F-2	146.7	119.8 rpm	14.8 rpm	0.123
4	F-2	146.7	119.4 rpm	14.8 rpm	0.124
5	F-2	146.7	119.8 rpm	15.0 rpm	0.125
9	F-2	146.7	126.0 rpm	16.0 rpm	0.127
10*	F-2	146.7	125.5 rpm	10.2 rpm	0.081
17**	Dummy	145.0	121.6 rpm	10.94 rpm	0.090

* One 46.5-gram shim was added to each despin weight for test number 10.
** One 34-gram shim was added to each despin weight for test number 17.

Theoretical calculations based on ideal conditions did not provide results identical to the test results. The differences between the two sets of results were attributed to inaccuracies in the data used in the theoretical calculations, and to the fact that the tests could not be conducted under simulated space conditions. Since it was not possible to precisely predict the "in space" spin-down ratio, RCA concluded that it would not be advisable to attempt to spin down to the optimum spin rate of 12 rpm. Instead it was decided to provide "safe-side" spin down and then, if necessary, to increase the spin rate by firing a pair of spin-up rockets. Accordingly, standard 47.7 gram shims were added to each of the despin weights on the flight model satellites.

The actual despin ratio (ω_f/ω_i) for the TIROS II satellite, as calculated from telemetered data, was 0.065. This ratio was slightly less than the theoretical ratio of 0.07, which had to be computed for the 47.7-gram despin weights.

3. TV Camera Response Tests

The TV camera response tests included checking the TV subsystems spectral response as well as its response to varying scene brightness. The spectral-response checks were made to determine the effect of the "non-visible" light spectrum on the vidicon output and to determine the differences in the spectral response of the individual vidicons. The brightness-response checks were made to ensure that the vidicon's threshold and saturation points were within the required limits.

The spectral-response checks were made after the TV subsystem was integrated into the satellite. To facilitate these checks, the camera shutters were removed, and the required sync pulses and regulated voltages were supplied by external sources. The checks were made using a monochrometer, a Perkin-Elmer 112V spectrometer, and an oscilloscope, which was used to measure the output of the camera electronics box. Figure 72 shows the plots of the spectral responses of the narrow-angle and wide-angle camera systems installed on satellite F-2. As can be seen from these plots, the "non-visible" light spectrum had little effect on the output from the TV subsystem.

The TV camera's response to a varying scene brightness was checked using two 500-watt lamps, of known characteristics, that were operated at 3200 degrees Kelvin. The outputs of the two TV channels were monitored as the light intensity was increased from 300 foot-lamberts to 10,000 foot-lamberts.* Plots of the brightness response of the wide angle and narrow-angle TV cameras of satellite F-2 are shown in Figures 73a and 73b, respectively.

4. Solar-Cell Output Tests

The solar-cell assemblies used on the TIROS II satellite were residual from the TIROS I project. Since these assemblies had been tested extensively during TIROS I, the TIROS II tests were limited to rechecks of the assemblies' overall operation. This recheck was

*The test-lamp characteristics and the intensity limits were established during meetings between RCA and the Weather Bureau.

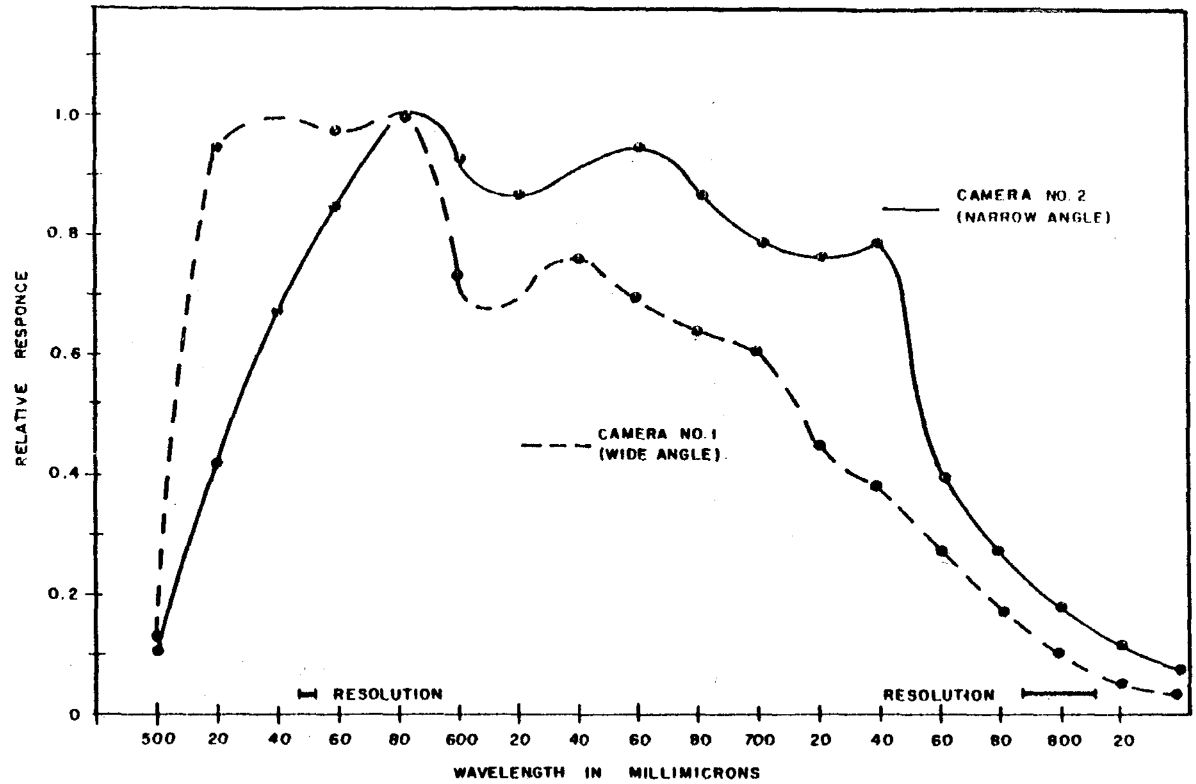


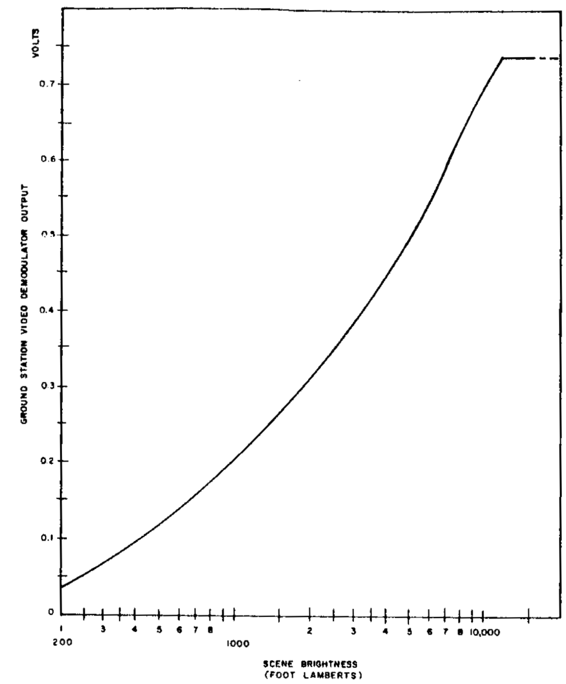
Figure 72. Spectral Response of TV Cameras

accomplished by shading various sections of each assembly while leaving other sections exposed to sunlight. The output voltage and current of each solar-cell section was measured for various load conditions. A pyroheliometer was used to measure the intensity of the light for each test. To facilitate this recheck procedure, the assembly was mounted in the special test fixture that had been developed during TIROS I. Figure 74 shows the plot of the outputs which were recorded when only the top of the assembly was exposed to sunlight.

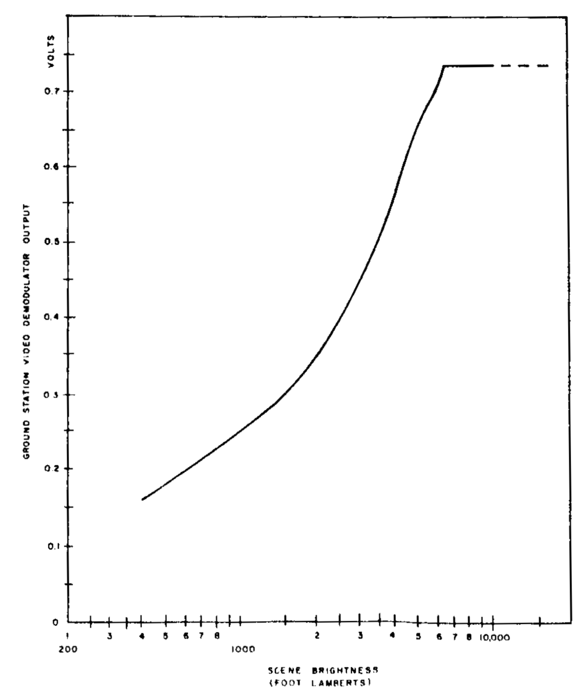
5. Thermal Tests

Thermal testing was limited to the battery racks and to new or modified subsystems. Since the items of equipment that were residual, or government furnished, from TIROS I had already passed the stringent TIROS I thermal-vacuum tests, they were not subjected to further subsystem level tests.

The tests, conducted in RCA's 48-inch thermal-vacuum chamber, were very similar to those performed during TIROS I. However, since the TIROS I telemetry data had shown that the battery temperature never exceeded 30 degrees centigrade, NASA issued a waiver that changed the temperature range within which the batteries were to be tested. Thus instead of being tested at -10 degrees and +60 degrees, the TIROS II battery packs were "soaked" at -10 degrees and +55 degrees, and were operated only within the temperature range of 0 degrees to +40 degrees centigrade.



a. Brightness Response of Wide-Angle Camera



b. Brightness Response of Narrow-Angle Camera

Figure 73. TV Cameras, Response to Varying Scene Brightness

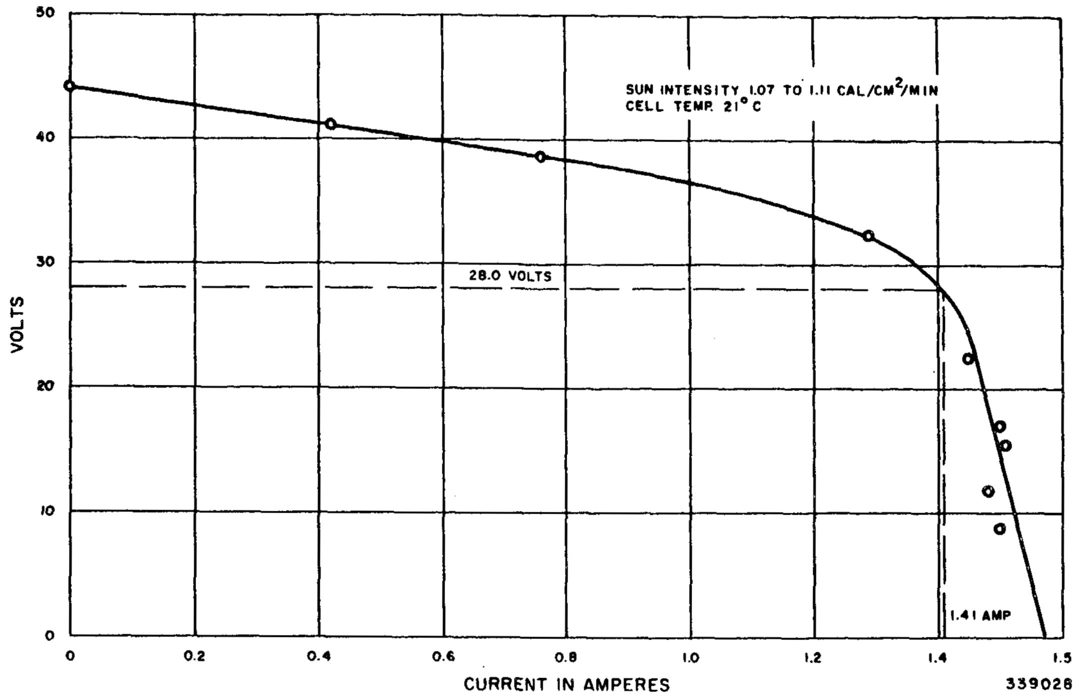


Figure 74. Plot of Output from Top-Mounted Solar Cells

The operation of the new and modified subsystems was tested within the temperature of -10 degrees and +60 degrees centigrade. When the testing was conducted at temperatures that were either less than -10 degrees or in excess of +60 degrees, operating power was supplied by a source other than the satellite batteries.

B. ENVIRONMENTAL TEST

1. Introduction

The TIROS II environmental tests were conducted to demonstrate that the TIROS satellite could survive the launch and orbit environments. The requirements for these tests were derived from information obtained from previous launchings of other satellites (including TIROS I), current specifications, and contractual agreements.

The probability of survival of a satellite is governed by: (1) the accuracy of predicting the expected environment, and (2) either the number of satellites tested or the severity of the tests relative to the predicted environmental conditions. Because of the limited number of TIROS satellites, the TIROS environmental testing program was aimed at subjecting the satellites, and their components, to conditions that were much more severe than those which would be encountered during launch and orbit. The prototype satellite was tested at levels which were up to three times greater than the predicted launch and orbit environments. Since the prototype satellite successfully passed the "high-level" tests, it was concluded that the design of the satellite and its subsystems was more than adequate. Therefore, the actual flight model satellites were subjected to tests which were at the same level as the predicted launch and orbit environment.

The TIROS II environmental test program consisted of the following phases:

a. Prototype Satellite Test. The T-2A prototype satellite was subjected to vibration, acceleration, spin, and thermal-vacuum tests. The level of testing was approximately three times as severe as the phase "C" test level.

b. Flight Satellite Component Tests. Each new component of the flight satellites was subjected to vibration, acceleration, and thermal-vacuum tests. These tests were designed to simulate orbital parameters. After the new components successfully passed this phase of environmental testing, they were integrated into the flight model satellites.

c. Flight Satellite Tests. The integrated flight model satellites were subjected to vibration, spin, and thermal-vacuum tests. The levels of the vibration, and spin tests were selected to simulate the anticipated launch-phase environment. The thermal-vacuum test provided an environment more severe than the orbital environment to which the satellite would be exposed.

These test phases are described in detail in Volume III, Appendix C, of the TIROS I Final Report (Reference 1).

2. Equipment and Performance Evaluation

The satisfactory performance of components under test was defined by the individual unit specification. The performance of completed satellites or sub-assemblies was evaluated by equipment similar to that of the ground station. Duty cycles of units were approximated during testing. The equipment, which would be operating during the launch phase, was operated and monitored during vibration and shock tests. All other equipment was turned off during the tests. The thermal-vacuum tests were long-term tests during which the units were usually operated at their nominal duty cycle. However, the duty cycles were sometimes increased to provide for accelerated life tests.

The first meeting of the TIROS II Environmental Test Program Committee resulted in the adoption of a modified version of RCA Specification TSP TI-100B. Portions of the specification relating to the satellite tests were incorporated into the TIROS procurement specifications which became the official test document.

During the TIROS program, the Environmental Test Program Committee met periodically to review the status of the testing program and resolve new problems. The solutions of the problems required certain additions, deletions, and changes to the specifications.

The Environmental Test Committee, on the basis of past experience, predicted that a large portion of system troubles would be due to the thermal-vacuum parameters. Therefore, considerable time was allocated to thermal-vacuum testing. Tests were performed in the

48-inch diameter environmental chamber at RCA-AED. External, rack-mounted, Go, No-Go test equipment, and the Princeton Ground Station equipment were used to monitor the performance of the satellite under test. The baseplate and side temperatures were measured and recorded by using copper-constantin thermocouples and a temperature-calibrated recording potentiometer.

SECTION IV. SATELLITE SYSTEM TESTS

A. BALANCING THE SATELLITE

The TIROS II satellites had to be statically balanced to within 20 ounce-inches, and had to be dynamically balanced to within 6 ounce-inches. These balance requirements were necessary because of the adverse effect that an excessive unbalance would have had on the spin axis of the combined satellite and launch vehicle. The maximum permissible unbalance was calculated as described in Volume I, Appendix J of the TIROS I Final Report (Reference 1).

Static balancing was the first phase of the balancing program. This static balance was achieved by distributing the total weight of the satellite so that the axis of rotation passed through the satellite's center of gravity (c.g.). Dynamic balancing of the TIROS II satellite was accomplished through use of the test fixture shown in Figure 75. This fixture allowed the satellite under test to be rotated at a constant velocity. When dynamic forces were detected in two support planes, the satellite was balanced by adding the necessary weights to overcome the unbalance.

Immediately before shipment to Cape Canaveral, Florida, each satellite was given a final dynamic balance. This final balance reduced the remaining unbalance in TIROS satellite F-2 to 3.4 ounce-inches in plane 1 (camera side) and 0.8 ounce-inches in plane 2 (top).

A detailed description of the test fixture and of the procedures employed in balancing a TIROS satellite is presented in VOLUME III of the TIROS I Final Report (Reference 1).

B. SATELLITE VIBRATION TEST

1. Summary

The series of vibration tests which were to be conducted on the TIROS II satellites were begun in late June, 1960. At that time, the No. 1 Infra-Red (IR) Can Response Test was started to determine the transient response of the IR package to a 1 g rms sine-wave input at frequencies between 20 and 2000 cps. Results of the test showed no adverse amplification factors in the thrust direction; however, an amplification factor of 25 (at 68 cps) was detected in the top of the package in one lateral direction. New support brackets were designed and installed on the IR package to dampen this amplification.

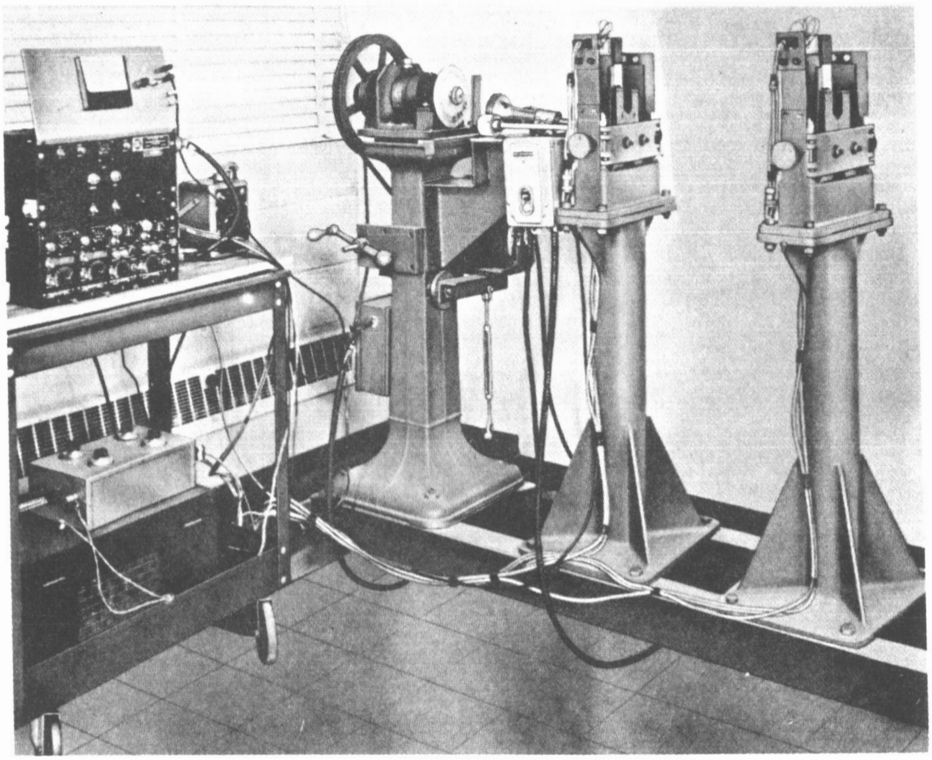


Figure 75. Dynamic Balancing Equipment

Prior to commencement of the No. 2 IR Can Response Test, studies were conducted to determine whether or not it would be necessary, or advisable, to repeat the entire vibration test. Several factors lead to the decision to repeat only those procedures which had not been successfully completed during the No. 1 test. Among these factors were: (1) results of analyses of the frequency response pattern showed that if the "worst" amplification factor were considerably lowered all other amplification factors would also be lowered; and (2) results of studies indicated that excessive, prolonged vibration testing of the package structure might lead to fatigue of certain members and thus might increase the possibility of structural failure.

On August 11, 1960, the No. 2 IR Can Response Test was performed in the 20-degree to 200-degree lateral direction. Results of this second test showed that the redesign of the brackets had resulted in a lowering of the amplification factor from the initial 25 at 68 cps to 10 at 90 cps.

The second phase of vibration testing was the preparation for the random noise third level test on satellite T-2A. Prior to running this test, the satellite's vibration response curve was plotted and the input levels required to equalize the system to within ± 3 db were determined. The third-level test was then run using these equalization levels. The test was as follows:

(a) Thrust Direction. 20 to 2000-cps bandwidth 20 g rms for two minutes ($0.2g^2$/cps power density).

(b) Two Lateral Directions. 20 to 2000-cps bandwidth, 14 g rms for two minutes.

Except for an overspeed indication on the IR speed control, and a few minor failures such as broken wires, the response of the T-2A satellite was satisfactory during this phase of testing.

On September 19, 1960, the third phase of vibration testing, the 600-cps 1000-pound force test, was run on satellites T-2A and F-4. Both satellites successfully passed this phase of the vibration test.

Flight vibration tests were started on satellite F-4 on September 30, 1960 and were successfully completed on October 1, 1960.

Satellite F-2 became eligible for flight vibration testing on October 9, 1960. The satellite successfully passed these tests on October 11, 1960.

2. Infra-Red Can Response Test No. 1

a. Objective

The objective of this test was to investigate the transient response of the infra-red (IR) package when subjected to a 1 g rms sine-wave input which was swept from 20 to 2000 cps. The test was conducted on prototype satellite T-2A.

b. Procedure

The IR package was monitored with accelerometers mounted in three mutually perpendicular planes, while the satellite was vibrated in the thrust direction and two radial directions. The test procedures were as follows:

(1) The accelerometers were calibrated by the displacement method.

(a) Oscillator frequency was calibrated using Lissajou's Pattern Method.

(b) A 10-1 scribe wedge was mounted on the vibration table for an optical displacement measurement.

(c) The output of the accelerometers was read on a Ballantine rms meter and as peak voltage on a calibrated Tektronix oscilloscope. The sensitivity of the accelerometers was such that they read 70 mv rms equivalent to 1 g rms.

(2) The IR package was fitted with an aluminum tube having a volume of approximately one cubic inch. The cube, mounted on the top of the IR package approximately one half-inch from the rim, had three accelerometers affixed to it. Figure 76 shows the arrangement of the accelerometers on the package.

(a) Accelerometer D (#3876) responded to any acceleration in the satellite thrust direction.

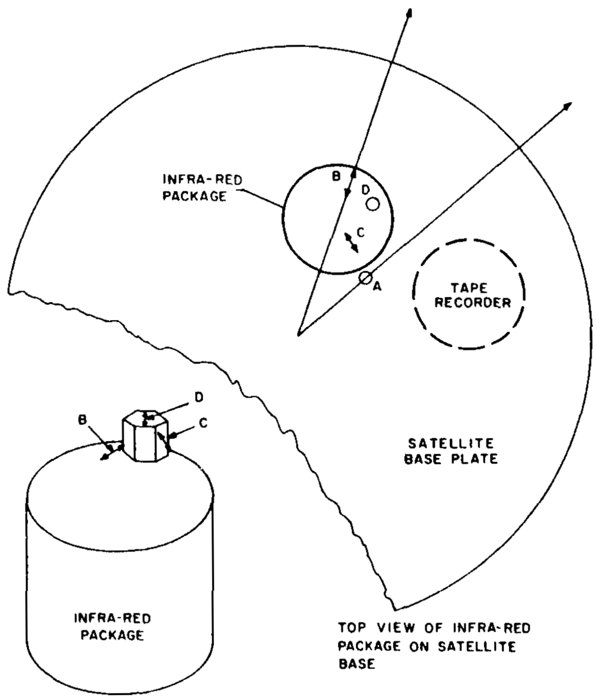


Figure 76. IR Can Response Test, Arrangement of Accelerometers

(b) Accelerometer B (#3874) was the radial reference accelerometer. It responded to any acceleration in a radial direction from the center of the satellite through the center of the IR package.

(c) Accelerometer C (#3884), which was mounted 120 degrees from the accelerometer B, responded to any acceleration at the top of the IR package in a 140 — 320 degree direction (called tangential in this section).

(d) Accelerometer A (#3892) was mounted in the satellite baseplate, about 6 inches from the center of the satellite, in a 50 degree direction.

(3) Initially the satellite was vibrated in the thrust direction by applying a 1 g rms sinusoidal input, swept from 20 to 2000 cps, for a 6 minute duration. The responses of the four accelerometers A, B, C, D, and the input accelerometer were plotted through a log pre-amplifier on the X-Y recorder. The results of this vibration are listed here:

Accelerometer	Frequency (cps)	Amplification Factor	db
A(#3892)	22	7	17
	48	30	29
	150	5	14
	400	6	16
	520	6	16
	820	5	14
B(#3874)	1000	2	6
C(#3884)	48	3	10
	66	6	16
	96	2	6
	150	3	10
	160	3	10
D(#3876)	1000	2	6
	1600	6	16

(4) The satellite was then vibrated in a lateral direction (20 — 200 degree axis) for a duration of six minutes. The data recorded during this second vibration is listed in the following tabulation:

Accelerometer	Frequency (cps)	Amplification Factor	db
A(#3892)	22	5	13
	45	9	19
	66	8	18
	130	6	15
	165	3	8

Accelerometer	Frequency (cps)	Amplification Factor	db
B(#3874)	67	14	23
	88	3	11
	120	6	16
C(#3884)	68	25	28
	86	11	21
	120	10	20
D(#3876)	67	7	17
	110	3	10
	140	6	15
	170	3	8

(5) The final vibration test made along the satellite's 140 — 320 degree axis, also had a six minute duration. The results of this vibration are tabulated below.

Accelerometer	Frequency (cps)	Amplification Factor	db
A(#3892)	22	6	15
	43	9	19
	46	9	19
	70	11	21
	80	11	21
	100	3	10
B(#3874)	22	2	4
	78	14	23
	160	3	9
C(#3884)	72	14	23
	105	10	20
	140	7	17
	170	3	11
D(#3876)	80	5	13

c. Conclusions

An analysis of the test results * lead to the following conclusions:

(1) The natural frequency of the IR package was 1600 cps.

*The reading obtained on accelerometer A, located on the satellite's baseplate, were not considered relevant to the vibrational response of the IR package and thus have no bearing on the test conclusions.

(2) The vibrational response of the IR package in the thrust direction was satisfactory (the maximum detected amplification factor was 6).

(3) A redesign of the IR package's dampening brackets would be necessary in order to decrease the amplification factor of 25 which was recorded on accelerometer C, while the satellite was being vibrated in the 20—200 degree lateral direction.

(4) The redesign of the dampening brackets, called for in (3), would not only reduce the maximum amplification factor but would also provide a general reduction of all the amplification factors.

3. Infra-Red Can Response Test No. 2

a. Objective

The objective of this test was to determine the IR package's response to a 1 g rms sinusoidal input whose frequency was being swept 20 to 2000 cps, and thus to determine the effectiveness of the newly designed dampening brackets.

b. Procedure

This procedure was limited to the 20 — 200 degree lateral direction. The decision to limit the test to these parameters came after an analysis showed that a decrease in the maximum detected amplification factor, 25 at 68 cps, would be accompanied by a general decrease in all of the amplification factors.

The accelerometers were calibrated and mounted as described in the procedures for IR Can Response Test No. 1. The satellite was then vibrated in the 20 — 200 degree lateral direction with a servo-controlled input of 1 g rms for ten minutes. While the frequency of the input was swept from 20 to 2000 cps, an X-Y recorder was used to plot the output from each of the accelerometers.

The following tabulation lists the results of this second test and includes a listing of the results of the first test to facilitate comparison:

	Amplification Factor	
Accelerometer	Test No. 1	Test No. 2
B(#3874)	14 at 67 cps 6 at 120 cps	10 at 90 cps 4 at 200 cps
C(#3884)	25 at 68 cps	10 at 90 cps
D(#3876)	7 at 67 cps	3 at 90 cps

c. Conclusions

The results of the test showed a general improvement in all measured amplifications factors and thus verified the design of the dampening brackets. Since this general improvement was noted, and since the "worst" case amplification factor had been reduced from 25 at 68 cps to 10 at 90 cps, it was concluded that the IR package's response in the other two directions would remain within limits and that additional testing would not be required.

4. 600-cps, 1000-Pound, Force Test

a. Objective

The objective of this test was to determine whether or not the satellite could withstand the 600-cps vibration which had been detected during several test firings of the third-stage rocket. The test procedures dictated that this check be made by vibrating the satellite with a 1000-pound force input while the vibrating frequency was swept from 550 cps up to 650 cps and then back to 550 cps.

b. Procedure

Prior experience had shown that in the test frequency range, the TIROS II satellite would present a complex impedance to the shaker table. Therefore, before the actual test could be commenced, it was necessary to plot the vibrational impedance of the satellite. This plot was then used to determine the number of "g's" input to apply to the satellite's "effective mass" or, in the event that the satellite acted as a spring*, the magnitude of the displacement that should be applied.

A force measuring fixture containing strain gages was used to link the satellite and the shaker table to facilitate readout of the force input to the satellite. The use of the strain gages on this fixture allowed a true force reading to be taken regardless of the type of impedance offered by the satellite.

A block diagram of the test set-up used for this phase of vibration testing is shown in Figure 77. A mobility-analog diagram of the system (satellite and test set up) is shown in Figure 78. An analysis of the diagram shows that regardless of the impedance of the tank circuit, the force input (current) is always equal to the current into the mass (C).

The force applied to the varying effective mass of the satellite was maintained constant by varying the velocity input (integrated accelerometer output). This was achieved by amplifying the strain-gage outputs and applying them to the oscilloscope and Solatron resolved-components indicator for phase comparison with the integrated accelerometer signal. The impedance characteristics of the satellite were plotted from the changes in phase relationship that occurred while vibrating frequency was swept from 550 to 650 cps.

*It was found that the TIROS I satellite acted as a spring in the 550- to 650-cps range, and it was anticipated that the TIROS II satellite would act in much the same manner.

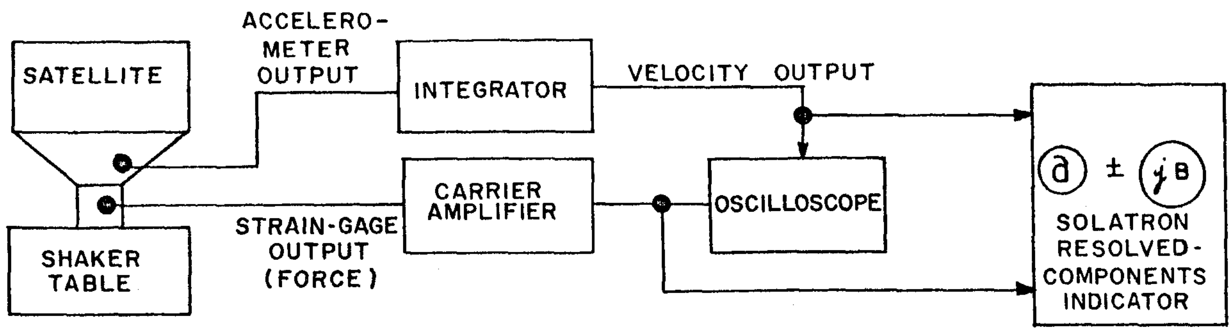


Figure 77. 600-CPS 1000-Pound Force Test, Test Setup

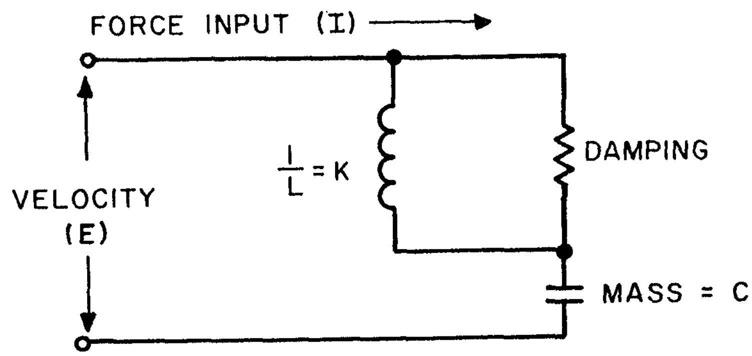


Figure 78. 600-CPS 1000-Pound Force Test, Mobility Analog Diagram

The impedance plot showed that the satellite acted as a spring in the 600-cps test range. Using this data, the "spring" constants were calculated and the deflections were converted to "g" values. When the actual vibration test was conducted, the "force" inputs were held constant at 500-pounds peak by varying the acceleration level between 30 and 50 g's. During the period of this test, the 550-cps to 650-cps frequency range was swept in both the upward and downward direction. Each sweep period was 30 seconds.

The results of this test conclusively showed that TIROS satellite T-2A could withstand the anticipated 600-cps vibration. Also, since the satellite acted as an effective spring, the 600-cps vibrational requirement was waived for the flight satellites.

5. Vibrational History of the Satellites

The following tabulations list the vibration test dates and test parameters for each of the TIROS II satellites.

TABLE 8. SATELLITE T-2A, VIBRATION TEST PROGRAM

Date	Run No.	Payload Position	Frequency Range (cps)	G's (rms)	Power Density (G^2/cps)	Duration of Run (minutes)
8-12-60	1	Vertical	20 to 800	13.0	0.20	2.0
8-17-60	2	Vertical	20 to 800	13.0	0.20	2.0
8-20-60	3	Vertical	800 to 2000	16.5	0.20	0.5
8-20-60	4	1st Lateral	20 to 800	9.2	0.10	2.0
8-24-60	5	1st Lateral	20 to 700	8.5	0.10	2.0
8-24-60	6	1st Lateral	700 to 1400	9.0	0.10	0.5
8-24-60	7	1st Lateral	1400 to 2000	8.0	0.10	0.5
8-25-60	8	2nd Lateral	20 to 700	8.5	0.10	2.0
8-25-60	9	2nd Lateral	700 to 1400	9.0	0.10	0.5
8-25-60	10	2nd Lateral	1400 to 2000	8.0	0.10	0.5

TABLE 9. SATELLITE F-1, VIBRATION TEST PROGRAM

Date	Run No.	Payload Position	Frequency Range (cps)	G's (rms)	Power Density (G^2/cps)	Duration of Run (minutes)
9-30-60	1	Vertical	20 to 450	3.2	0.025	2.0
9-30-60	2	Vertical	450 to 2000	6.2	0.025	0.5
10-1-60	3	1st Lateral	20 to 800	4.6	0.025	2.0
10-1-60	4	1st Lateral	800 to 2000	5.5	0.025	0.5
10-1-60	5	2nd Lateral	20 to 740	4.3	0.025	2.0
10-1-60	6	2nd Lateral	750 to 2000	5.6	0.025	0.5

TABLE 10. SATELLITE F-2, VIBRATION TEST PROGRAM

Date	Run No.	Payload Position	Frequency Range (cps)	G's (rms)	Power Density (G^2/cps)	Duration of Run (minutes)
10-9-60	1	Vertical	20 to 715	4.2	0.025	2.0
10-10-60	2	Vertical	700 to 2000	5.7	0.025	0.5
10-10-60	3	1st Lateral	20 to 500	3.5	0.025	2.0
10-10-60	4	1st Lateral	500 to 1000	3.6	0.025	0.5
10-10-60	5	1st Lateral	1000 to 2000	5.0	0.025	0.5
10-11-60	6	2nd Lateral	20 to 450	3.6	0.025	2.0
10-11-60	7	2nd Lateral	450 to 900	3.6	0.025	0.5
10-11-60	8	2nd Lateral	900 to 2000	5.5	0.025	0.5

TABLE 11. SATELLITE F-4, VIBRATION TEST PROGRAM

Date	Run No.	Payload Position	Frequency Range (cps)	G's (rms)	Power Density (G^2/cps)	Duration of Run (minutes)
9-19-60	1	Vertical	20 to 500	3.5	0.025	2.0
9-22-60	2	Vertical	20 to 500	3.5	0.025	2.0
9-23-60	3	Vertical	500 to 2000	6.0	0.025	0.5
9-23-60	4	1st Lateral	20 to 450	3.2	0.025	2.0
9-24-60	5	1st Lateral	400 to 950	3.8	0.025	0.5
9-24-60	6	1st Lateral	900 to 2000	5.4	0.025	0.5
9-24-60	7	2nd Lateral	20 to 550	3.8	0.025	2.0
9-24-60	8	2nd Lateral	500 to 950	3.5	0.025	0.5
9-24-60	9	2nd Lateral	950 to 2000	5.3	0.025	0.5

C. STANDARD PERFORMANCE – EVALUATION TEST

The standard performance evaluation test is described in the classified supplement to this report.

D. QUALIFICATION TESTS

The qualification tests, or acceptance tests, were performed in accordance to contractual agreements. These tests consisted of a series of both environmental and standard performance-evaluation tests which were arranged so as to provide a means of demonstrating the satellite ability to withstand the anticipated launch and orbital conditions. Environmental tests were monitored closely, and complete electrical calibration checks were made at the conclusion of each test. The calibration checks were designed to uncover any changes in satellite operation due to environmental testing. Prior to shipment to Cape Canaveral, Florida, each satellite was given a final calibration check, its sensors were carefully aligned, and it was both statically and dynamically balanced.

The satellite qualification tests consisted of the following sequences of environmental and standard performance-evaluation tests. The actual test procedure is presented in Volume III of the TIROS I Final Report (Reference 1).

1. The standard performance-evaluation test (described in Appendix B of the Classified Supplement of this report).
2. A spin test. The spin test conducted only on the prototype satellite, consisted of spinning the satellite about its longitudinal axis at 200 rpm and operating the components that would be operational during launch. The test duration was 500 seconds.
3. A de-spin test (performed on the prototype model only).
4. A check of the satellite's magnetic dipole moment.
5. Alignment of the TV cameras, and the IR and horizon sensors.
6. The standard performance-evaluation test.
7. A vibration test. The level at which the prototype was vibrated was higher than the level at which the flight-model satellites were vibrated. Power was applied only to those components that would be operative during the launch phase; namely, the beacon transmitters, the command receivers, and the clocks and their oscillators.
8. The standard performance-evaluation test.
9. A thermal-vacuum test. This test permitted monitoring of the satellite's operation in a simulated orbital environment. The pressure within the thermal-vacuum chamber was maintained between 5×10^{-5} mm Hg and 1×10^{-4} mm of Hg. The prototype satellite was tested at temperatures of -10 and +60 degrees centigrade; the flight model satellites were tested at 0 and +50 degrees

centigrade. During these tests, the satellite was interrogated at two hour intervals.* Each interrogation consisted of the following commands and programs:

a. Direct Camera Sequences for both camera chains

b. Playback Sequence for both camera chains

c. Set clocks

d. Start clocks

e. Beacon kill and restart

10. The standard performance-evaluation test.

*When the satellite was operated at temperatures below 0 degrees or above +50 degrees centigrade, satellite power was supplied from an external source. The reasons for this are presented in Section IV of this part of the Report.

E. FINAL CHECK BEFORE SHIPPING TO CAPE CANAVERAL

1. General

At the conclusion of the qualification test program, each satellite was subjected to a rigorous examination and then made ready for shipment to the launch site at Cape Canaveral, Florida. This final check consisted of both an electrical test and a mechanical inspection. The mechanical inspection was performed to ensure that all screws and nuts were tight, that all electrical connections were secure, and that the satellite was free from foreign material, such as dust and dirt. The electrical test was basically a repeat of the standard performance-evaluation test (described in Appendix B of the Classified Supplement to this report). However, the test also included sealing each adjustable component into its proper position, measuring the satellite's magnetic dipole moment, checking the camera sensitivity and focus, and aligning the sun sensors and IR sensors.

After completion of the electrical and mechanical checks, the satellite was rebalanced, its moment of inertia was measured, and its final weight was checked. When balancing was completed and the moment of inertia was established, the satellite was subjected to an additional electrical check and then prepared for shipment. The type of shipping container in which the satellite was transported to the Cape Canaveral launch site is shown in Figure 79.

Figure 79. TIROS II Shipping Container

2. Measuring the Satellite's Magnetic Dipole Moment

The following is a tabulation of the magnetic dipole moments of TIROS II flight model satellite F-2. The test apparatus and test procedures used in measuring the satellite's dipole moments is described in the Design and Development Part of this report.

TABLE 12. MAGNETIC DIPOLE MOMENTS OF SATELLITE F-2

Operational Mode	Magnetic Dipole Moment (Ampere-Turns-Meter^2) Orbital Day	Orbital Night
Standby	+0.10	-0.31
Remote-Picture taking with both cameras in sequence	+0.42	
Playback-Camera 1	+0.08	-0.53
Playback-Camera 2	+0.05	-0.62
Direct 1	+0.26	
Direct 2	+0.11	

3. Alignment and Calibration of the TV Cameras

a. Objectives

The objectives of the TV camera alignment and calibration procedure were to: (1) ensure that the optical axis of the TV camera did not deviate from being parallel to the satellite's spin axis by more than one degree; (2) obtain "calibration" negatives and prints by taking pictures of field-targets with each camera system; and (3) determine the "field-of-view" distortion of each camera system. The TIROS II procedure was somewhat different than the TIROS I procedure in that the field-targets were located on an outdoor range and were located further from the TV cameras than the TIROS I targets. These changes were made to provide a better check on the focusing of the satellite's TV cameras. The distances between targets was computed by RCA and verified by the National Bureau of Standards.

The satellites had to be dynamically balanced prior to the start of the TV camera alignment and calibration procedure.

b. Alignment of Camera Axis

The alignment test fixture is illustrated in Figure 80. The satellite was mounted in a horizontal position with its balancing shaft on ball bearings. The front surface of mirror A was mounted at the end of the balancing shaft, perpendicular to the spin axis. Two crosshairs, centered on the satellite's spin axis, were inscribed on the mirror. The centering of the crosshairs was achieved by rotating the satellite and shaft, and positioning the mirror so that the intersection of the crosshairs appeared to be stationary.

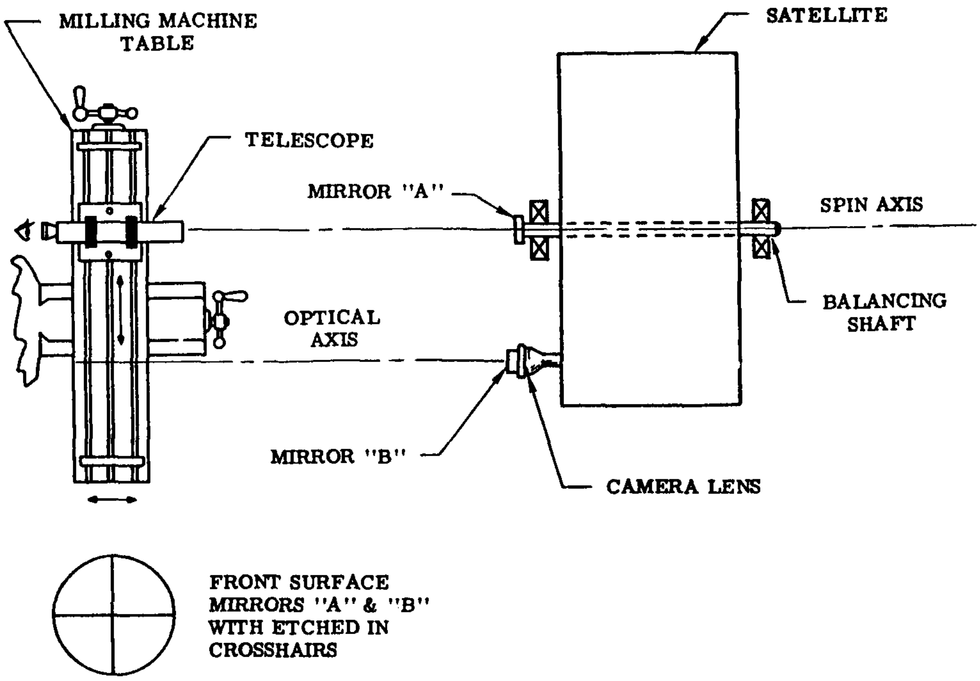


Figure 80. Test Fixture for Alignment of TV Cameras

A Taylor Hobson telescope was mounted on a specially calibrated milling machine table. The milling table was arranged so that it was perpendicular to the satellite's spin axis.

The satellite was placed a convenient distance from the alignment telescope with the TV cameras facing the satellite. Parallelism and coincidence between the telescope optical axis and the satellite spin axis was achieved by focusing the telescope for twice the distance between its front element and the surface of mirror A, and noting the reflection of the telescope target in the mirror.

Mirror B, also with crosshairs inscribed at right angles on its surface, was placed on the front element of the camera lens and the mirror was adjusted so that the center of the crosshairs was coincident with the optical center of the lens. The alignment telescope was then moved along the milling machine table (pure translation) until the built-in crosshairs of the alignment telescope were coincident with the crosshairs of mirror B on the lens surface. The deviation from parallelism of mirror B, with respect to the satellite spin axis, was determined (by simple triangulation) by focusing the alignment telescope for twice the distance between the front element of mirror B and noting the reflection of the telescope target in the mirror as viewed through the telescope.

To check the accuracy of the measurement, the telescope was moved back, without further adjustment of focus, etc., so that it faced the center of mirror A. Since the telescope optical axis remained parallel to the spin axis of the satellite throughout the test, the deviation noted while the telescope was aimed at mirror B represented the angular deviation of the optical axis of the wide angle camera with respect to the satellite spin axis.

This same procedure was used to determine the deviation from parallelism of the optical axis of the narrow angle camera with respect to the satellite's spin axis. The data for these measurements is contained in Reference 5. The TIROS satellites met the requirement of less than one degree deviation between the satellite's spin axis and the optical axis of each of the two TV cameras.

c. Photographing Camera Calibration-Targets

The satellite was placed in its test fixture on a platform, and positioned so that the TV cameras faced the test targets. The general test setup is depicted in Figure 81.

Mirror B (Figure 82) was mounted on the front element of the wide angle camera lens and the center of the mirror crosshairs was made coincident with the optical center of the lens. The mirror surface was adjusted to be perpendicular to the optical axis of the lens. Mirror B faced toward target A, which was located 60 inches from the front nodal point of the wide angle camera lens.

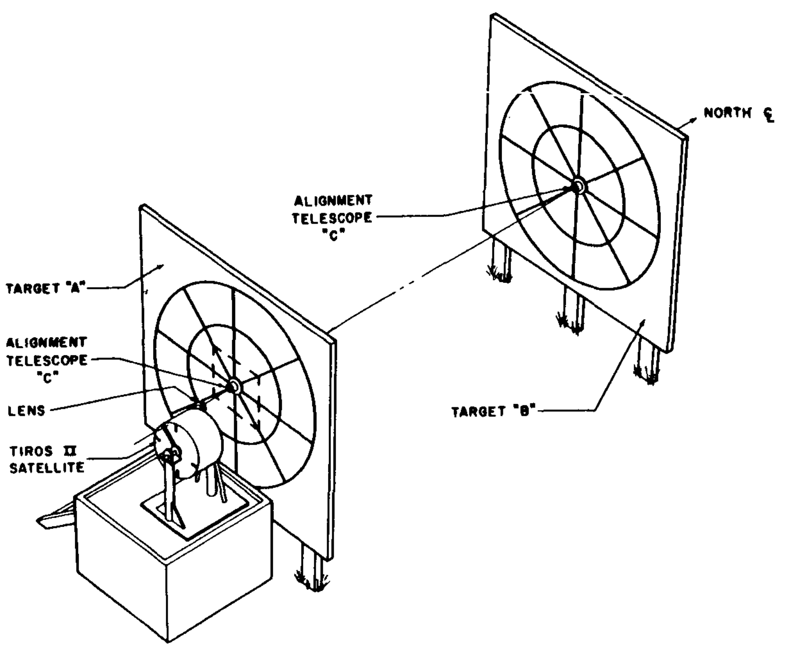


Figure 81. Test Setup for Calibration of TV Cameras

A Taylor Hobson alignment telescope, C, was mounted through the center of target A. The mounting of the telescope assured perpendicularity between the telescope optical axis and the front surface of target A. (As noted previously, the telescope was equipped with built-in crosshairs that could be "zeroed-in" on the optical axis of the oscilloscope.)

With the crosshairs of the telescope in the "zeroed-in" position, mirror B was viewed through the telescope. The satellite was positioned so that the crosshairs on mirror B and the crosshairs of the telescope appeared coincident when viewed through the telescope. The optical axis of the lens was thus centered with the center of the target. Perpendicularity between the camera-lens optical axis and the front surface of target A was achieved by switching on the telescope's collimating light and noting the reflection of the telescope target in mirror B as viewed through the telescope.

After the lens optical axis was centered on target A, and after the axis was made perpendicular to that target, mirror B was removed. After removal of the mirror, a series of pictures of target A was taken through the wide angle camera system of the satellite.

The same procedure was used to align the narrow angle camera to target B. The distance from the lens front nodal point to the front surface of target B was 50 feet (nominal).

Since target B was placed behind target A, a special door was cut in the center of target A. When the door was opened, target B could be viewed by the narrow angle camera. The Taylor Hobson telescope was placed in the center of target B for this alignment procedure.

Typical test photographs of the calibration targets are contained in Reference 4, as is other pertinent alignment data.

4. Alignment of the IR Sensors

a. Objectives

The objectives of the IR sensor alignment procedure were to ensure that the sensitive axis of the five-channel radiometer intersected the satellite's spin axis at an angle of 45 ±0.25 degrees and that the sensitive axis was within ±0.25 degrees of lying along (or being parallel to) the satellite's 350-degree radial line.

b. Procedures

The alignment procedure was accomplished using the test setup shown in Figure 82. The satellite was positioned so that its spin axis (AB) was parallel to the milling-machine table (EF) and perpendicular to the line-of-sight (BC) of the alignment oscilloscope. The satellite was then rotated until the 350-degree radial line was in the same horizontal plane as the spin axis and the line-of-sight.

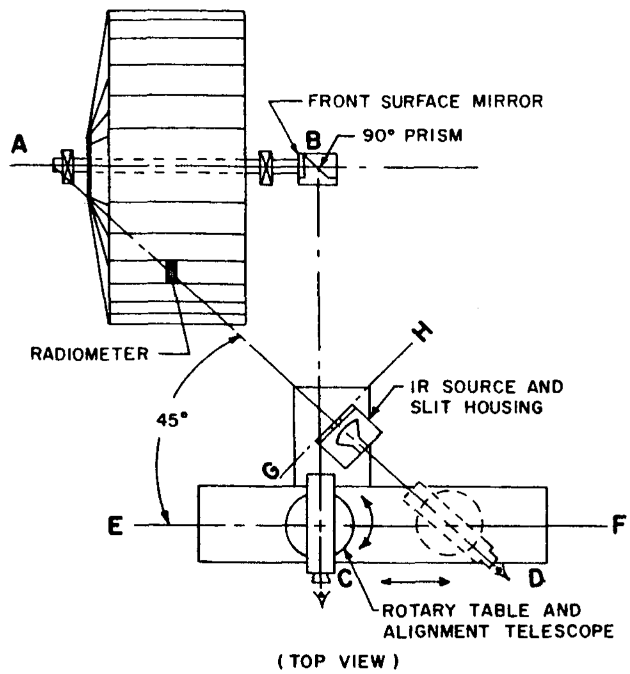


Figure 82. IR Sensor Alignment Apparatus

A front-surface mirror was installed on the "B" end of the satellite shaft as was a 90-degree pentaprism. The alignment telescope was affixed to a rotary table, calibrated in degrees, which in turn was mounted on the milling table. The telescope was set-up so that it was perpendicular to EF (and AB) and so that the mirror's crosshairs were brought into view. The rotary table was then adjusted so that the telescope's line-of-sight made an angle of 45 degrees with EF. The rotary table was then moved along EF until the telescope's line-of-sight (AD) was aligned with the center of the AD prism.

In order to check the vertical field-of-view of each sensor, a plate with a vertical slit was mounted so that it could be tracked in a direction GH, perpendicular to AD, and an infra-red heat lamp was mounted "behind" the slit.

The radiometer was put into operation, and the IR source (the lamp and plate) was moved along line GH. When the output from the radiometer reached a maximum, the IR source was locked into position and the IR heat lamp was removed from the IR source. After the lamp was removed, the telescope was sighted on the radiometer prism by combined rotation and translation. The angular displacement of the telescope from the 45-degree reference was recorded as the deviation in the IR channel sensitive axis. This procedure was repeated for each of the five radiometer channels.

The half-power points for each radiometer channel were than measured, using a similar technique. The results of the sensitive-axis (vertical field-of-view) alignment check, which was conducted on flight model satellite F-2, are tabulated here.

TABLE 13. SATELLITE F-2, ANGLE-OF-RESPONSE POINTS (WITH RESPECT TO SPIN AXIS)

Radiometer Channel	Half-Power Point	Peak	Half-Power Point	Average of Half-Power Points
1	41°40'	45°25'	45°40'	43°40'
2	45°00'	45°40'	46°40'	45°30'
3	41°00'	45°40'	46°00'	43°30'
4	43°40'	45°25'	47°40'	45°40'
5	41°30'	44°00'	47°00'	44°20'

The horizontal field-of-view of the sensors was measured using procedures similar to those used to determine the vertical field-of-view. The primary difference was that the plate-slit was positioned horizontally. The measurement, which was conducted only on radiometer channel 3, showed that the sensor's horizontal field-of-view deviated from the 350-degree radius by 2.55 x 10^{-3} radians in the 349-degree radius direction.

5. Alignment of the Attitude Indicator Subsystem

a. Requirements

The sensitive axis of north-indicator reference sensor had to be in the vertical plane which passed through the spin axis and the 340-degree reference line on the satellite's baseplate. The sensitive axes of the remaining eight sensors had to be evenly spaced about the periphery of the satellite so that the angular displacement of each axis was a multiple of 40 degrees (±45 minutes) when referenced to the 340-degree reference sensor.

The sensitive axis of the horizon scanner had to lie in the plane which intersected the satellite's spin axis at an angle of 90 degrees ±15 minutes. It was also required that this sensitive axis be oriented radially with respect to the spin axis. (The accuracy of this requirement was limited to several degrees since the scanner lens had to be in position to "peer" through the window provided in the satellite's skin.)

b. Procedures

The procedures used in the alignment of the TIROS II north indicator and horizon scanner are the same as the procedures described in Volume III of the TIROS I Final

Report (Reference 1). Reference 3 (Alignment and Calibration Data for the TIROS II Meteorological Satellite System) lists the results of the alignment procedures. These results show that all of the alignment requirements were met.

6. Final Balance of the Satellite

The final-dynamic-balance procedure was very similar to the initial balance procedure described elsewhere in this report. This final balance was performed after the satellite "hat" was secured to the satellite baseplate. Removal and replacement of the "hat," or any other components, after the final balance was completed would have voided the balance.

The final balancing efforts on flight model satellite F-2 were successful in reducing the remaining unbalance to 3.4 ounce-inches in Plane 1 (Camera Side) and 0.8 ounch-inches in Plane 2 (Top). This remaining unbalance was well within the maximum unbalance limit of 6 ounce-inches. The weights which had to be added to achieve this minimum unbalance were either cemented to the top surface with epoxy cement, or screwed into the flanges on the satellite's baseplate.

7. Determining the Satellite's Moments of Inertia

The satellite's moment of inertia about its spin axis was determined while the satellite was mounted in the balancing machine in a horizontal position, with a 1-inch diameter shaft for support, and a weight of approximately 5 pounds was bolted to the top cover by means of the lifting-lug tapped hole. After the preliminary procedures were completed, the satellite was displaced from its center position and allowed to oscillate as a compound pendulum. The period of the system oscillations was recorded approximately ten times and an average value was computed from the recorded data. The equation which was used to compute the satellite's moment of inertia about its spin-axis, in terms of this average period, is as follows:

$$J = W_r L \left(\frac{T^2}{4\pi^2} - \frac{L}{g} \right)$$

where

J	Mass Moment of Inertia, lb-inch-sec^2
L	Length of Pendulum, inches
W_r	Reference Weight, pounds
T	Period of Oscillations, seconds

The satellite's moment of inertia about an axis normal to the spin axis (transverse moment of inertia) was determined by attaching the satellite to a Bifilar suspension system and measuring the period of the resultant pendulum. The measurements were made for four separate orientations of the satellite. The first measurement was made while the satellite was positioned so that its 110-degree radial line was at the top vertical (the point nearest to the pendulum's pivot). Subsequent measurements were made with the 155-degree, 200-degree, and 245-degree radial lines were at the top vertical. The transverse moment of inertia (for each orientation of the satellite) was then computed using the following equation:

$$J = \frac{Wd^2T^2}{16\pi^2 L}$$

where:

J	Mass Moment of Inertia, lb-inch-sec^2
L	Length of Pendulum, inches
W	Weight of Test Body, pounds
T	Period of Oscillation, seconds
d	Distance between Bifilar Cables, inches

A more detailed description of procedures and test apparatus used in determining the satellite's moments of inertia is presented in Volume III of the TIROS I Final Report (Reference 1). The final moment of inertia about the spin axis of flight model satellite F-2 was 153.5 lb-inch-sec^2. The transverse moments of inertia of satellite F-2 are tabulated below.

Satellite Orientation Angle	Transverse Moment of Inertia (lb-inch-sec^2)
110^o	107.2
155^o	105.6
200^o	101.8
245^o	103.0

F. CHRONOLOGICAL HISTORY OF THE SATELLITES

1. Prototype Satellite T-2A

Prototype satellite T-2A (designed prototype T-2A during TIROS I) was designated to meet the rigid non-flight test requirements. Testing of the satellite was commenced on June 27, 1960.

During the initial standard performance evaluation test, a fuse opened in the 24.5-volt regulator. The cause of failure was traced to regulator No. 3, which was subsequently replaced with regulator No. 8. Upon completion of the standard performance-evaluation test, T-2A was sent back to the system integration group for installation of the infra-red (IR) equipment, wiring modifications, calibration, and the correction of minor difficulties. After the maintenance was completed, the satellite was subjected to IR Can Response Test No. 1. During this test an amplification factor of 25 to 68 cps was detected. Accordingly, modifications were made to the support brackets for the IR package before IR Can Response Test No. 2 was conducted. The results of this second test verified the bracket modifications in that the maximum amplification factor was reduced from 25 to 68 cps to 10 at 90 cps.

On August 15, 1960, a functioning IR subsystem was delivered by NASA and installed on the T-2A baseplate. The standard performance-evaluation test was then conducted. The results of this test indicated that all of the satellite's subsystems, including the IR subsystem (checked by NASA), operated normally.

On August 17, 1960, T-2A was dynamically balanced, subjected to a second standard performance-evaluation test, and then prepared for vibration testing. Random noise vibration (20 to 2000 cps) having an amplitude of 20 g rms was applied in the thrust direction for 2 minutes. During this vibration test, the tapes on both TV tape recorders were dislocated, a screw was sheared in a supporting bracket (not considered a functional failure), and a wire in the filament and focus circuit of the vidicon was broken, causing a failure in the No. 2 TV camera system. The dislocation of the tapes in the TV tape recorders was attributed to a test procedure error. (The recorders were vibrated in the record position rather than in the playback position as required.) After the satellite was repaired, it was again subjected to the vibration test.

During the second vibration test, a malfunction occurred in the No. 2 TV camera system (wide-angle camera). This malfunction was attributed to a defective tantalytic capacitor. Investigation disclosed that the capacitor failure could not be attributed to the vibration test in that the capacitor was firmly imbedded in place and no evidence of mechanical rupture existed. After the necessary repairs were made, the satellite was subjected to random noise vibration, from 800 to 2000 cps, at an amplitude of 16.5 g rms. This vibration was applied in thrust direction for a duration of two minutes. The satellite checkout, which was conducted at the conclusion of this test, indicated an overspeed of the IR package's motor-speed control unit. This overspeed occurred for approximately 2/3 of the initial playback period, but did not occur during any other playback period.

The next phase of vibration testing was the application of a 20-cps to 800-cps random vibration in the lateral plane. This vibration, which had an amplitude of 9.2 g rms, was applied for a period of one minute. Post-vibration checkout of the satellite revealed a broken wire in the No. 2 TV tape recorder, a variation in synchronization of TV channel No. 2, and the recurrence of the overspeed in the IR motor-speed control unit. The cause of the variation in synchronization was traced to a slipping belt in No. 2 TV tape recorder and subsequently was corrected. The IR package was removed from the satellite to facilitate determination of the cause of the intermittent overspeed. Shortly after removal, a broken lug was discovered on the back of an electrical connector. An investigation showed that the lug had probably been damaged prior to "potting." Therefore, it was concluded that the vibration had aggravated rather than caused the failure.

On August 23, 1960, the environmental test committee reviewed the results of vibration testing and recommended that:

a. Random vibration (20 to 2000 cps) be applied in the lateral direction to ensure against the recurrence of troubles that had been encountered during earlier vibration testing.

b. The TV tape recorder wires be fastened more securely; that each epoxy connection in the flight model satellites be rechecked; and that all other wire connections be carefully reinspected.

c. All belt tensions and flight-model TV recorders be rechecked.

On August 25, 1960, the IR package was reinstalled on the satellite's baseplate, and satellite T-2A was again subjected to random vibrations in the lateral plane (20 to 700 cps at 8.5 g rms for 2 minutes, 700 to 1400 cps at 9.0 g rms for 0.5 minutes, and 1400 to 2000 cps at 8.0 g rms for 0.5 minutes). During these vibrations, the belt on the No. 2 TV tape recorder was dislocated. After the belt was repositioned, the test was completed. Further investigation of the belt system indicated that the high "runout" of the motor shaft was the primary cause of tape dislocation.

The next series of tests were the thermal-vacuum tests. During the initial test of this series, at a temperature of +25 degrees centigrade and a pressure of 5×10^{-5} mm of Hg, synchronization in the No. 2 TV tape recorder was lost as was the response of the IR radiometers. The loss of synchronization was again traced to a slipping belt. The gross loss of response of the radiometer was traced to improperly aligned targets in the vacuum chamber. However, one channel of the radiometer did contain a non-operating bolometer and the IR subsystem had to be removed from the satellite. The subsystem was repaired by NASA personnel and returned to RCA-AED September 8, 1960. Initial checkout of the unit revealed an intermittent sensor. However, since this malfunction did not affect the overall subsystem performance, the environmental test committee decided to continue the thermal-vacuum tests.

On September 10, 1960, thermal-vacuum testing was resumed. The pressure within the chamber was lowered to 5×10^{-5} mm Hg and the temperature was set to 0 degrees centigrade. Testing was continued at these levels until September 12, 1960, when the temperature was raised to +50 degrees centigrade. On September 14, 1960, the test was discontinued because of failures in the tape recorder of the No. 1 TV camera channel (narrow angle), the clock of the No. 2 camera channel (wide angle), and the radiometer section of the IR subsystem. The belt on the No. 1 TV tape recorder was thrown from the reel just as the belt had been thrown from No. 2 TV tape recorder during a previous test. After an investigation revealed that the reel webs in the recorder were cracked (probably as a result of damage incurred during an acceleration test) the recorder was replaced. The failure in Channel 4 and 5 of the radiometer was traced to a failure in the radiometer amplifiers and a defective radiometer sensor. These failures were corrected, and on September 20, 1960, the prototype satellite was subjected to further thermal-vacuum tests. The thermal-vacuum test continued at 0 degrees centigrade until September 23, 1960; on that date the temperature was raised to +50 degrees centigrade.

On September 26, 1960, the test was discontinued because: (1) the incremag and a filter resistor in the No. 2 clock failed; (2) the output transistor in the No. 1 clock failed; and (3) the IR subsystem failed. On October 5, 1960, after all necessary repairs had been made, the prototype satellite was subjected to 25 g's of acceleration in the thrust direction for three minutes. Post acceleration checkout revealed no mechanical or electrical failures.

Thermal-vacuum testing of satellite T-2A was resumed on October 8, 1960. On October 17, 1960, the tests were completed and the results were reviewed by the environmental test committee. The test committee recommended that additional thermal-vacuum tests be conducted at Fort Monmouth, New Jersey. The tests included one day at 0 degrees centigrade, three days at +50 degrees centigrade, and one-half day at +60 degrees centigrade. After completing these thermal-vacuum tests successfully, satellite T-2A was subjected to the final standard performance-evaluation test, dynamically balanced, aligned and calibrated, and prepared for shipment to Cape Canaveral.

2. Flight Model Satellite F-1

On September 27, 1960, Flight Model Satellite F-1 was received for testing from the systems integration group. On September 30, 1960, after completion of the initial balancing procedures and the spin test, satellite F-1 was subjected to random vibration along the thrust axis (20 to 450 cps at 3.2 g rms for 2.0 minutes and 450 to 2000 cps at 6.2 g rms for 0.5 minutes). A BNC connector was loosened from the diplexer as a result of these tests.

After replacing the BNC connector the satellite was subjected to the following random vibrations:

a. Lateral Plane No. 1

   (1) 20 to 800 cps at 4.6 g rms for 2 minutes

   (2) 800 to 2000 cps at 5.5 g rms for 0.5 minutes

b. Lateral Plane No. 2

   (1) 20 to 740 cps at 4.3 g rms for 2 minutes

   (2) 750 to 2000 cps at 5.6 g rms for 0.5 minutes

During post-vibration checkout, a broken wire was discovered in the wide-angle radiometer. However, an investigation showed that this condition had existed prior to vibration and, therefore, it was not considered to be a test failure. Repair of the radiometer was accomplished by NASA personnel. After the IR subsystem was reinstalled, the standard performance-evaluation test was successfully completed.

Thermal-vacuum tests were started on October 4, 1960. While the chamber temperature was being held at 0 degrees centigrade, an excessive level of standby current was detected. A defective voltage regulator was found to be the cause of trouble and was replaced. Testing was resumed on October 6, 1960. The test was discontinued on October 11, 1960, when a failure occurred in the IR package. The defective package was subsequently replaced with IR package No. 1.

On October 17, 1960, the Environmental Test Committee reviewed the test status of satellite F-1. As a result of this meeting, it was agreed to continue thermal-vacuum testing for three days at 0 degrees centigrade and for four days at +50 degrees centigrade, and to repeat the low-band, thrust-direction, vibration test. It was also agreed that the TV subsystem should be operated for only two of the three 0-degree days because of previous accumulative test time on that subsystem.

Thermal-vacuum testing was resumed on October 19, 1960. During the initial phase of this test, it was noted that the end-of-tape pulse was not being received. After NASA personnel corrected the cause of malfunction, the pressure within the thermal-vacuum chamber was again reduced. The test was stopped when it was noted that the video signals, the IR signals, and the operation of clock No. 2 were marginal.

On October 27, 1960, after all necessary repairs and adjustments were completed, thermal-vacuum testing was resumed and was continued until November 7, 1960. During this test interval a switch failed in the IR package. NASA personnel corrected this defect and testing was resumed on November 10, 1960. On November 13, 1960, difficulty was encountered in setting the satellite clocks and the satellite was removed from the thermal-vacuum chamber. The operation of the satellite was returned to normal by replacing the auxiliary control unit and the IR package, and by repairing the 32-volt regulator.

Thermal-vacuum testing of F-1 was again commenced on November 16, 1960. On November 18, 1960, a fuse opened in the 26-volt unregulated power supply. The cause of the blown fuse was determined to be an oversize stand-off terminal which shorted to a capacitor case. The proper stand-off was installed, the erase head in the tape recorder was cleaned, and the test was resumed. The test was discontinued on November 21, 1960, after failures occurred in clock No. 2 and the beacon-kill system. Although the necessary repairs were made, testing was not resumed because of the successful launch of satellite F-2.

3. Flight Model Satellite F-2

Flight Model Satellite F-2, which was later designated as the satellite to be launched, was received by the system test group from the systems integration group on October 8, 1960.

After the standard performance-evaluation test, the satellite was subjected to random vibration along the thrust axis (20 to 715 cps at 4.2 g rms for 2 minutes). After satisfactorily passing this phase of the vibration test, satellite F-2 was subjected to a thrust-axis vibration of 700 to 2000 cps, at 5.7 g rms, for 4 minutes. The satellite was then subjected to lateral plane vibration as follows:

a. Lateral Plane No. 1

(1) 20 to 500 cps at 3.5 g rms for 2.0 minutes

(2) 500 to 1000 cps at 3.6 g rms for 0.5 minutes

(3) 1000 to 2000 cps at 5.0 g rms for 0.5 minutes

b. Lateral Plane No. 2 (90 degrees from Plane No. 1)

(1) 20 to 450 cps at 3.6 g rms for 2.0 minutes

(2) 450 to 900 cps at 3.6 g rms for 0.5 minutes

(3) 900 to 2000 cps at 5.5 g rms for 0.5 minutes

Checkouts between each step proved that the satellite had passed the test satisfactorily. However, the post-vibration standard performance-evaluation test revealed that TV transmitter No. 2 had a varying output. Accordingly, the transmitter was replaced with a flight spare. (The cause of trouble was later traced to a faulty vacuum tube.)

Thermal-vacuum testing was commenced on October 15, 1960. On the second day of testing, the IR package malfunctioned and the test was discontinued. The cause of the malfunction was corrected and the thermal-vacuum test was resumed on October 18, 1960. The test was continued until October 20, 1960. On that date, the IR package again malfunctioned and the satellite had to be removed from the thermal vacuum chamber. The cause of failure was attributed to a defective diode and a loose wire in the R3-1 box of the IR package. Before testing was resumed, the defects were corrected, an RF filter was added to the input of the IR experiment, and a faulty shutter in the No. 1 TV camera was replaced.

The thermal-vacuum tests were resumed on October 22, 1960. On October 26, 1960, the RF output from the "E" antenna failed and the test was interrupted. When the thermal-vacuum chamber was opened, it was determined that the test cable had become disconnected from the satellite's antenna ring. The cable was reconnected and the test was completed without further incident. Satellite F-2 was then subjected to the final standard performance-evaluation test, dynamically balanced, aligned and calibrated, and prepared for shipment to the launch site at Cape Canaveral, Florida.

The satellite was shipped from RCA-AED on November 5, 1960. Upon arrival at its destination, the satellite was removed from its sealed, pressurized (dry nitrogen at 3 psig) shipping container; subjected to a Go, No-Go electrical check; and calibrated. After the satellite was approved by the NASA project manager, it was delivered to the spin test facility for balancing, mating with the third-stage rocket, and alignment.

The final weight of satellite F-2, as measured at RCA-AED, was 277 pounds 7-1/2 ounces. The weight check made by Douglas Aircraft Company (DAC) at Cape Canaveral, Florida, showed that the actual satellite weight (after correction for the difference in gravitational constant) was within 1.6 ounces of the weight which had been measured at RCA-AED. The center-of-gravity of the satellite was determined to be 10.062 inches above the separation plane. The final unbalance of the satellite was 3.4 ounce-inches in plane 1 (camera side) and 0.8 ounce-inch in plane 2 (top). The moment of inertia about the satellite's spin axis was 153 inch-pound-seconds^2.

On November 16, 1960, the satellite was erected on the service tower and prelaunch interrogations were begun. During the initial interrogations, the collimator on the narrow-angle camera inadvertently touched and shorted an external current limiter, causing a fuse in the power-supply circuit to open. The fuse was replaced without removing the satellite from the tower; the time required for the repair was less than 1.5 hours. After the fuse was replaced, the satellite was successfully interrogated.

Daily interrogations of the satellite were made without further incident until T-1 day. During the initial T-1 day interrogation, clock No. 2 alarmed 20 seconds early. However, the alarm time was correct for subsequent interrogations on that date. On T-0 day, clock No. 2 again alarmed 20 seconds early during the initial interrogation; but it alarmed normally on three additional interrogations. It was concluded, therefore, that spurious signals were responsible for the error in alarm time on both days. The command frequency was carefully monitored during all other interrogations with the result that no further difficulties were encountered in the clock circuits.

Shortly, prior to the launch of TIROS II, the NASA doppler facility determined the beacon frequencies to be as follows:

Beacon 1	107.999993 Mc
Beacon 2	108.027094 Mc

Flight model satellite F-2 was successfully launched at 0613.03.8 A.M., EST, November 23, 1960. The satellite was ejected into a nearly circular orbit having an apogee of 453 statute miles, a perigee of 387 statute miles, and a calculated eccentricity of 0.007.

4. Flight Model Satellite F-4

Satellite F-4 was the first flight model satellite to be tested. Upon receipt by the test group, the satellite was tested to determine the magnetic dipole moment developed by the wiring; the TV cameras were aligned; the satellite was balanced and subjected to a spin test; and a standard performance-evaluation test was performed. After the satellite was aligned and tested, it was subjected to vibration.

Initially the satellite was subjected to random vibration along the thrust axis (20 to 400 cps at 3.5 g rms) for 2 minutes. Post-vibration checkout of the satellite revealed that the IR subsystem was not functioning properly. Subsequent analyses showed that the tape recorder heads had become magnetized and required degaussing, and that a wire in the tape transport had broken.

Vibration testing was resumed on September 22, 1960, and was completed on September 24, 1960. During that period, the satellite was subjected to the following:

Direction	Frequency Range (cps)	Amplitude (g's rms)	Duration (minutes)
Vertical Plane	20 to 500	3.5	2.0
	500 to 2000	6.0	0.5
Lateral Plane No. 1	20 to 450	3.2	2.0
	400 to 950	3.8	0.5
	950 to 2000	5.4	0.5
Lateral Plane No. 2	20 to 550	3.8	2.0
	500 to 950	3.5	0.5
	950 to 2000	5.3	0.5

Thermal-vacuum testing was started on September 29, 1960. On Octorber 2, 1960, the playback function of the IR experiment failed and the test had to be interrupted. The IR experiment was removed from the satellite and returned to NASA for repair. The thermal-vacuum test program for satellite F-4 was allowed to remain in this interrupted status in order that all available time could be allotted to completing the tests on the other flight model satellites.

PART 4. FIELD OPERATIONS

SECTION I. PRINCETON GROUND STATION

The Princeton ground station was the prototype for the primary ground stations at the Pacific Missile Range and Fort Monmouth. Although the basic station was government-furnished equipment (residual from TIROS I), certain modifications were made to allow the testing of improvements which were recommended for the TIROS II system. The primary modifications were made in the TV-FM demodulator, the tape and computer control, the sun-angle computer, and the timing system. After being fully tested and approved at the Princeton station, these modifications* were incorporated in the two primary ground stations.

The Princeton station was also used to test the TIROS II satellites before they were shipped from the RCA-AED facility to the Cape Canaveral launch site. After launch of the TIROS II satellite, the Princeton station was used for backup of the primary station at Fort Monmouth. While operating in this capacity, the Princeton station recorded and plotted all telemetry and attitude data, and recorded and processed all TV pictures. At any time this station could have been used to program the satellite merely by applying plate voltage to the command transmitter.

Originally, the Princeton station was to serve in this backup capacity only during the first 48 hours after launch. However, on November 24, 1960, NASA directed that the station be maintained in its backup status until November 28, 1960. This directive was made so that the RCA engineering staff could participate more directly in the evaluation of the TIROS II satellite's initial performance. Of primary interest during this evaluation period was the erratic performance of the magnetic attitude control (MAC) stepping switch and the degraded quality of the pictures taken by the wide-angle camera system.

*The modifications are described in detail in Part 2, the Development and Design portion of this book.

SECTION II. CAPE CANAVERAL SUPPORT

A. PRELAUNCH

TIROS II launch site activities were essentially the same as those for TIROS I. RCA's activities at the Cape Canaveral Missile Test Area (CCMTA) were commenced on November 1, 1960. Prior to that date, representatives of NASA, USAF, RCA, and Douglas Aircraft Company (DAC) met to discuss coordination of the overall launch-site effort.

The prototype satellite, the required test equipment, and the Go, No-Go van were shipped from the RCA-AED facility near Princeton, New Jersey, on November 1, 1960, and arrived at CCMTA on November 3, 1960. An inspection and checkout of the satellite, Go, No-Go van, and test equipment revealed no signs of shipping damage. The satellite was then installed in the dust-free room (residual from TIROS I) and the Go, No-Go van was placed in position.

The checkout receiving and transmitting antennas, also residual from TIROS I, were remounted on the roof of Hangar AA and cabled into the Go, No-Go van and the satellite test area. An additional antenna, supplied by NASA, was used during checkout of the IR equipment. After the antennas were installed, the following signal strength measurements were made using signal generators, frequency counters, and etc.

1. Simulated command signals transmitted from Hangar AA and received at Launch Pad 17A.
2. Simulated command signals transmitted from Hangar AA and received at the spin-test building in Area 5*.
3. Simulated beacon, TV, and IR signals transmitted from Launch Pad 17A and received at Hangar AA.
4. Simulated beacon, TV, and IR signals transmitted from the spin-test building in Area 5* and received at Hangar AA.

After the simulated-signal measurements were made, the prototype satellite was interrogated first in the Area 5 spin building and then on the service tower at Launch Pad 17A. The interrogations were made with: (1) fairings on, tower in place; (2) fairings off, tower in place; (3) fairings on, tower removed; (4) fairings off, tower removed. The prototype satellite responded normally to all interrogations. Also, a comparison of the interrogation results with the simulated-signal measurements showed that the TIROS subsystems were operating within their specified power levels.

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* Auxiliary antennas were installed on the exterior of the spin-test building because the building was of steel construction.

Subsequent to the successful interrogations, the prototype satellite was returned to Area 5 for preliminary mating with the third-stage rocket and an alignment check. The alignment of the spin table and rocket-satellite combination was very close to nominal. Balancing drills, which were conducted using temporary weights attached to the third-stage rocket, showed that there would be no difficulty in meeting the balance requirements.

Flight-model satellite F-2, shipped from RCA-AED on November 5, 1960, arrived at the CCMTA on November 6, 1960. The satellite was checked and calibrated, and the calibration data was presented to the NASA project manager for approval. After approval of this data, the flight-model satellite was taken to the spin-test facility for mating with the third-stage rocket, alignment, and balancing. The runout of the spin table, rocket, and satellite combination was 0.0225 T.I.R. The final dynamic balance of this combination was achieved through use of approximately 300 grams of balance weights which were secured to the third-stage rocket with epoxy. The center-of-gravity of the satellite was determined to be 10.062 inches above the separation plane. The balance and the moments of inertia of the satellite were checked and found to be in close agreement with the measurements made immediately before the satellite was shipped from RCA-AED.

On November 16, 1960, DAC erected flight model satellite F-2 on the service tower. During satellite interrogations on that date, the collimator for the narrow-angle camera inadvertently touched and shorted an external current limiter, causing a fuse in the power-supply circuit to open. The fuse was replaced, while the satellite was installed on the tower, and the satellite was successfully interrogated.

Daily interrogations of the satellite were made without further incident until T-1 day. During the initial T-1 day interrogation, clock No. 2 alarmed 20 seconds early. However, the alarm time was correct for subsequent interrogations on that date. On T-0 day, clock No. 2 again alarmed 20 seconds early during the initial interrogation; but it alarmed normally on three additional interrogations. It was concluded, therefore, that spurious signals were responsible for the error in alarm time on both days. The command frequency was carefully monitored during all other interrogations with the result that no further difficulties were encountered in the clock circuits.

At T-9 minutes, the NASA doppler facility determined the beacon frequencies of the TIROS II satellite to be as follows:

Beacon 1	107.999993 Mc
Beacon 2	108.027094 Mc

B. LAUNCH

Successful launch of the TIROS II Meteorological Satellite occurred at 6:13 A.M. EST on November 23, 1960. The satellite was ejected into an orbit having an apogee of 453 statute miles, a perigee of 387 statute miles, and a calculated eccentricity of 0.007. The initial spin rate of 120 rpm was reduced to 8 rpm by successful operation of the satellite's de-spin mechanism. (On November 25, 1960, two pairs of spin-up rockets were fired, in response to commands from the back-up CDA station at the RCA-AED facility, to increase this "safe-side" spin rate to 14 rpm.)

After the successful launch of the TIROS II satellite, the prototype satellite, the Go, No-Go equipment, and the other test equipment were returned to the RCA-AED facility. The check-out antennas were removed from their mounting areas and placed in a NASA-supplied storage facility.

SECTION III. WASHINGTON, D.C. CONTROL CENTER

The TIROS II Control Center was located at the Goddard Space Flight Center in Greenbelt, Maryland (near Washington, D.C.). Shortly before launch of the TIROS II satellite, two RCA-AED engineers and three RCA Service Company Representatives were assigned to the Control Center. The engineers set up a training program to familiarize the Service Company personnel with the TIROS concept and with Control Center operations. The operating personnel had to be trained to do the following:

1. Accept, analyze, and catalog engineering reports on satellite and ground station condition.
2. Accept, analyze, and transmit predictions as to the sequence and locations of feasible photographic areas, as well as the times at which the satellite could be contacted by each ground station.
3. Accept daily programming recommendations from the United States Weather Bureau.
4. Decide the actual sequence of operations for each day, taking into consideration the condition of the power supply and the performance of the satellite and ground stations.
5. Prepare and transmit specific operating instructions and pertinent acquisition and picture orientation data to the ground stations.
6. Accept and review performance reports and significant telemetry data from the command and data acquisition stations.
7. Select and transmit immediate instructions in case of emergency.
8. Coordinate all TIROS II operational-phase activities.

The RCA-AED engineers remained at the Control Center as technical advisors until approximately six weeks after launch of the TIROS II satellite. After the RCA Service Company personnel were thoroughly familiar with Control Center operations, the engineering staff was reassigned to another phase of the TIROS Project. However, the staff was available, at any time, for technical assistance in the event of non-routine or emergency operations.

SECTION IV. PACIFIC MISSILE RANGE

A. LOGISTICS OF MOVE FROM KAENA POINT, HAWAII

Scheduling conflicts with other satellite-tracking operations limited the usefulness of the Kaena Point location as a TIROS CDA station. Because of this, it was decided that the TIROS II CDA station should be located at a different site; namely, the Pacific Missile Range. The actual moving operation was commenced shortly prior to June 30, 1960. The equipment vans were shipped by commercial movers from the ground station site to Honolulu, Hawaii. The vans were transported by ship from Honolulu to Los Angeles, California, and then flown by the USAF to Mc Guire Air Force Base, New Jersey. One van arrived at the Air Base on July 12, 1960; the second van arrived on July 14, 1960. The vans were then transported by commercial movers to the RCA-AED facility near Princeton, New Jersey.

After the vans were modified for use in the TIROS II system, acceptance tests were conducted to demonstrate the operability of the equipment to NASA. (The tests are described in Part 3, Section I.) On September 8, 1960, after acceptance of the equipment by NASA, the vans were shipped by commercial movers to Dover, Delaware. The equipment was flown from Dover to the Pacific Missile Range by the USAF.

B. EQUIPMENT INSTALLATION

The Pacific Missile Range (PMR) CDA station was divided between two sites. All the receiving, command, IR, attitude, telemetry, and data tape recording equipment, and some of the TV equipment was located on San Nicolas Island (SNI), one of the channel Islands, located about 55 miles from the California mainland and approximately equi-distant from Los Angeles and Santa Barbara. The remainder of the ground-station equipment was located at the Naval Air Missile Test Center, Point Mugu installation, on the mainland coast about 40 miles northwest of Los Angeles.

The station was divided in this manner because the AT-36 antenna, required for signal reception, was already constructed on SNI, whereas the facilities required for the photographic processing and meteorological interpretation of the TV film data were in existence at Point Mugu.

For the SNI installation, the two TIROS equipment vans were placed adjacent to each other and a short distance from telemetry building 182 which housed the control and tracking equipment for the AT-36 antenna system. The vans were joined by a short, enclosed walkway to facilitate movement of personnel and test equipment. A trailer was located alongside the vans to provide office and storage space and to house the NICOLA (designation for SNI installation) teletype facility. The layout of equipment at PMR is shown in Figure 83.§

§ This illustration is printed on a foldout page located at the rear of this Section.

The TV subsystem readout facility located at Point Mugu consisted of three racks of equipment. During the first four months of operation, these were installed in a house trailer similar to the one on SNI. The small amount of space remaining in the van was used for storage and office activity.

The nearby building 552 was the center for all meteorological activity at Point Mugu. Film strip projectors and other facilities located on Point Mugu were used by the TIROS meteorological team for immediate analysis of TV subsystem data. The PMRWEA (designation for PMR Weather Bureau facility at Point Mugu) teletype installation was also located there. Following completion of additions to building 552 in March 1961, the TV readout racks were moved from the trailer to space in building 552. The extensive photographic processing facilities, already in existence at Point Mugu, were utilized for development of the 35-mm data film positives for immediate analysis. In addition, a prefabricated, transportable, photographic darkroom was maintained for the processing of test and calibration films.

Communication of data and incidental traffic between the SNI and Point Mugu sections of the TIROS installation was accomplished by use of the PMR Instrumentation Data Transmission System (IDTS) microwave link. This system was installed and maintained during the TIROS II operational period by Collins Radio Company personnel. Signals from the SNI vans were transmitted via cable to telemetry building 182 and then were transmitted to another building on SNI via a short microwave link. The principal microwave link then relayed the signals to Point Mugu via an intermediate station on Santa Cruz Island.

The only breadboard IDTS channel required was a 150-kc band for transmission of the TV subcarrier and sun-pulse subcarrier from SNI to Point Mugu. In addition, audio channels were provided for two-way voice intercommunications, command programmer "source" tones from SNI to Point Mugu, and shutter actuation indicating signal from Point Mugu to SNI. The voice channels tied together "squawk-box" units in the 3 SNI trailers, the Point Mugu trailer, and buildings 182 and 552. The "source" tones were demodulated and used to indicate wide or narrow angle camera and direct or tape-playback modes to the display equipment. The shutter actuation signal was recorded on the Esterline-Angus events recorder on SNI.

The TIROS teletype installations at Point Mugu building 552 (PMRWEA) and at the SNI office trailer (NICOLA) were essentially identical. At each station two machines were provided, one each connected to NASA circuits AY 111 and AY 1806. The AY 111 circuit included the Minitrack network stations and the resultant heavy message traffic limited TIROS usage primarily to administrative messages. Consequently, the AY 1806 circuit was used primarily for long data messages. The NICOLA and PMRWEA installations used machines built by Teletype Corporation which were capable of transmitting from keyboard or punched tape and of typing hard-copy of received messages. Punched tape could be generated from a keyboard input but not from another tape or incoming message.

Telephone service was provided at PMR by two independent systems. A "white ball" administrative phone was installed in the SNI office trailer and at the Point Mugu station location. This phone reached other on-base "white-ball" extensions, as well as long distance service via the Point Mugu switchboard. Three additional phones on the "red-ball" test-communications network were installed, one in each SNI van and one in the SNI office trailer. These could only be used to call other "red-ball" phones at PMR.

The AT-36 antenna used for TIROS operation on SNI was installed shortly before the TIROS II launch. Calibration and acceptance tests were not performed at that time owing to the absence of boresight towers; consequently, the actual antenna parameters were unknown.

The TV and IR signals were picked up by separate vertical and horizontal probes at the antenna pods, filtered, amplified and conducted to building 182 via the dual rotary joint in the antenna pedestal. Two multicouplers then fed the signals to two receivers in building 182 (required for auto-track operation) and the four receivers in CDA Van 2.

The two command transmitters associated with the PMR CDA station were installed in the pedestal of the AT-36 antenna on SNI. The RF output cable was looped loosely from the top of the pedestal (to allow for antenna motion) and the other end secured to the dish structure and connected to the command antenna. This antenna was a Yagi type with ground-plane screen, and was mounted on the top rim of the 60-foot AT-36 antenna with its axis parallel to the dish axis.

C. TRAINING

The personnel training program at PMR was instituted during the equipment installation and training period, and was in effect until just prior to the launch of the TIROS II satellite from Cape Canaveral, Florida. The program, consisting of both classroom instructions and actual "on-the-equipment" training, was under the direction of two RCA-AED engineers and two RCA Service Company technicians. A total of 12 equipment operators were trained during this program.

Classroom instructions included discussions on the theory of operation and the operating procedures for the TIROS II ground stations. The on-the-equipment phase of the training program afforded the opportunity for the operating personnel to become familiar with equipment operating and maintenance procedures. The "Instruction and Operating Handbook, TIROS II Meteorological Satellite System" and the system schematics were used as reference material for the training program.

Simulated command programs were issued in the standard format and dry runs were made so that the operating personnel could gain experience in programming the satellite. All phases of programming were set up and checked against stop watch and events recorder records.

Test tapes of composite IR data were played through the quick-look demodulator to provide the operators with experience in that playback function. The calibrator unit, a signal generator, and pre-recorded tapes were used to train personnel in the operating procedures for the TV receiving and processing circuits of the ground station. A device that simulated horizon scanner information was used, along with a noise generator, to familiarize personnel with the operation of the attitude pulse selector and the digital time-measuring device.

To familiarize operating personnel with routine maintenance, the RCA representatives introduced "faults" into the system, and also guided the operators in adjusting the equipment.

The one-month delay in the launch of TIROS II, afforded the operators the chance to perfect their skills. On the date of launch, the operating personnel at the launch site consisted of four shift supervisors and eight operator technicians.

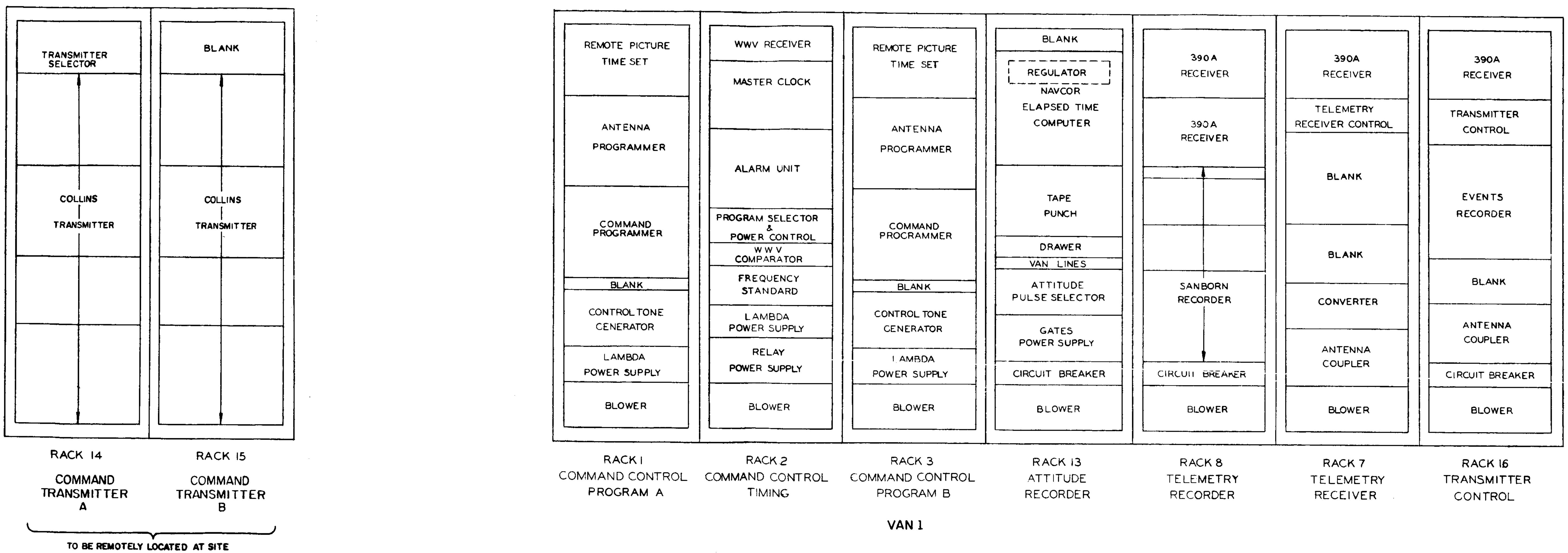
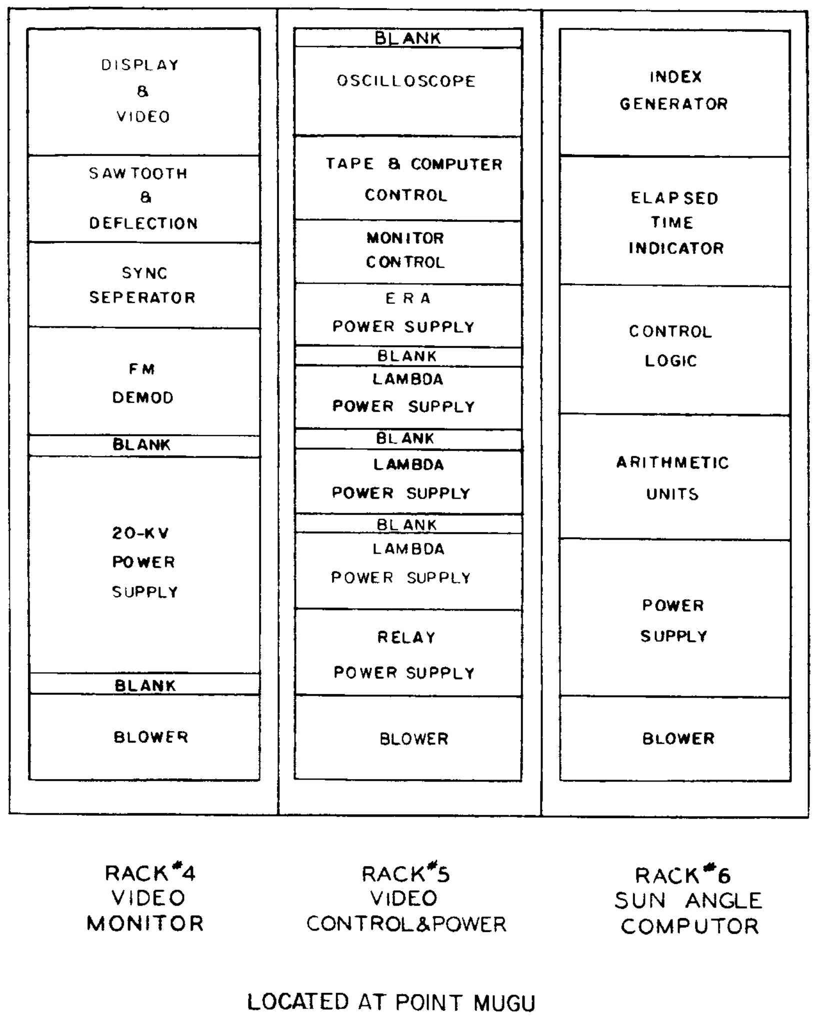


Figure 83. Pacific Missile Range, Layout of Equipment



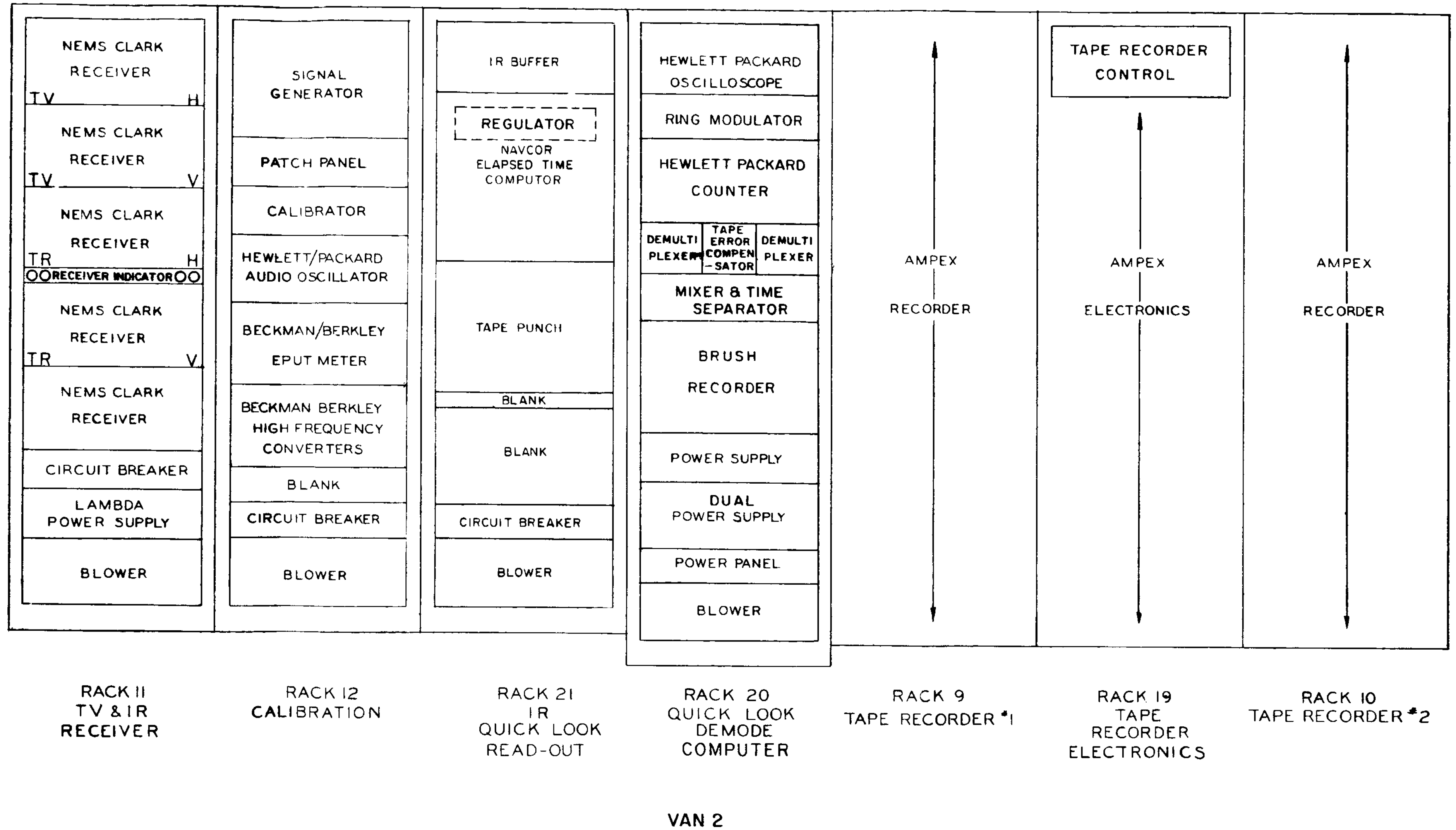


Figure 83. Pacific Missile Range, Layout of Equipment (Continued)

SECTION V. FORT MONMOUTH

The basic Fort Monmouth ground station was residual (government furnished) from TIROS I. However, certain improvements were made and certain additional equipment was installed prior to the launch of the TIROS II satellite. These modifications and improvements are described in Part 2, Section III of this report.

A change was also made in the operating personnel for the station. Although the Signal Corps maintained the operating responsibility for the tracking station and the maintenance responsibility for the tracking antenna, station operating personnel were all from the RCA Service Company. Engineering support was supplied by RCA-AED. The equipment racks of the Fort Monmouth ground station are shown in Figure 84.

Figure 84. Fort Monmouth Ground Station

PART 5. REFERENCES AND APPENDICES

REFERENCES

1. "Final Comprehensive Technical Report, TIROS I Meteorological Satellite System," RCA-AED 339, July 1, 1961.

2. "Operational Procedures for Power Supply Monitoring for the TIROS II Meteorological Satellite."

3. "Instructions Book for 1412 and 1432 Telemetry Receivers," Nems Clarke Co., Silver Spring, Md.

4. "Instruction and Operating Handbook TIROS II Meteorological Satellite System," October 1, 1960.

5. "Alignment and Calibration Data for the TIROS II Meteorological Satellite."

6. "Static and Dynamic Electricity," William Smythe, Second Edition, McGraw-Hill Book Co., New York, N. Y.

APPENDICES

APPENDIX A. EQUATIONS USED FOR PREDICTING THE PRECESSION OF THE SATELLITE'S SPIN AXIS

LIST OF SYMBOLS

S, $\bar{S}$	Satellite spin-axis vector
N, $\bar{N}$	Satellite-orbit normal vector
Θ	The anomalistic angle in the orbit plane from the ascending node to the projection of the sun vector onto the orbit plane
R	Magnitude of radius vector
Δt	Iteration interval
I	Satellite's moment of inertia about its spin axis
J	Satellite's moment of inertia about its spin axis
$\dot{\theta}_s$	Angular velocity of satellite along orbit
ω	Angular velocity of satellite about its spin axis
i	Inclination of orbit
Ω	Right ascension of ascending node
F	Factor proportional to magnetic dipole moment of satellite
V_o	Earth's surface magnetic field
C_ω	Spin-rate decay constant

LIST OF SUBSCRIPTS

x, y, z	Earth inertial coordinate frame; orthogonal right-handed; x toward first point of Aries; z along earth's spin axis
d	Day
n	Night
r	Residual

EQUATIONS OF PRECESSION

$$G_x = \frac{\epsilon}{2} \cos \rho \; (S_y N_z - S_z N_y)$$

$$G_y = \frac{\epsilon}{2} \cos \rho \; (S_z N_x - S_x N_z)$$

$$G_z = \frac{\epsilon}{2} \cos \rho \; (S_x N_y - S_y N_x)$$

$$H_x = \mu \; (S_y b_z - S_z b_y)$$

$$H_y = \mu \; (S_z b_x - S_x b_z)$$

$$H_z = \mu \; (S_x b_y - S_y b_x)$$

$$\dot{S}_x = G_x + H_x$$

$$\dot{S}_y = G_y + H_y$$

$$\dot{S}_z = G_z + H_z$$

$$S_{x_{i+1}} = S_{x_i} + \dot{S}_x \Delta t$$

$$S_{y_{i+1}} = S_{y_i} + \dot{S}_y \Delta t$$

$$S_{z_{i+1}} = S_{z_i} + \dot{S}_z \Delta t$$

Where:

$$\frac{\epsilon}{2} = \frac{3}{2} \left(\frac{I - J}{I} \right) \frac{\dot{\theta}_s^2}{\omega}$$

$$\mu b_x = 3/2\,\mu_d \sin i \cos i \sin \Omega - \frac{3}{8\pi}(\mu_d - \mu_n) \sin i$$

$$\Big\{[2\theta_2 - 2\theta_1 + \sin 2\theta_1 - \sin 2\theta_2] \cos i_s \sin \Omega +$$

$$[\cos 2\theta_2 - \cos 2\theta_1] \cos \Omega \Big\}$$

$$\mu b_y = 3/2\,\mu_d \sin i \cos i \cos \Omega - \frac{3}{8\pi}(\mu_d - \mu_n) \sin i$$

$$\Big\{[\cos 2\theta_2 - \cos 2\theta_1] \sin \Omega -$$

$$[2\theta_2 - 2\theta_1 + \sin 2\theta_1 - \sin 2\theta_2] \cos i_s \cos \Omega \Big\}$$

$$\mu b_z = 3/2\,\mu_d\ (\sin^2 i - 1) + \frac{3}{8\pi}(\mu_d - \mu_n) \sin i$$

$$[2\theta_2 - 2\theta_1 + \sin 2\theta_1 - \sin 2\theta_2] -$$

$$\frac{(\mu_d - \mu_n)}{4}(2\theta_2 - 2\theta_1)$$

And

$$\mu_d = \frac{MF_d}{\omega} = M\,(F_{r_d} + F_i)/\omega$$

$$\mu_n = \frac{MF_n}{\omega} = M\,(F_{r_n} + F_i)/\omega$$

$$M = \frac{V_o}{IR_s^3}$$

$$\theta_1 = \odot + \pi\psi$$

$$\theta_2 = \odot - \pi\psi + 2\pi$$

$$\psi = 0.5 + \frac{1}{\pi}\left\{\tan^{-1}\left[\frac{(R_e/R_s)^2 - 1}{(\vec{L}\cdot\vec{N})^2 - (R_e/R_s)^2}\right]^{1/2}\right\}$$

when

$$(\vec{L}\cdot\vec{N})^2 - (R_e/R_s)^2 \leqq 0;\ \psi \equiv 1$$

$$\cos\rho = (\vec{S}\cdot\vec{N})$$

$$\omega_{i+1} = \omega_i - C_{\omega}\,\omega_i\,\Delta t$$

APPENDIX B. MAGNETIC DIPOLE MEASURING APPARATUS, THEORETICAL CALCULATIONS

The magnetic field within a spinning, uniformly-charged, spherical shell is constant in magnitude and parallel to the spin axis of the sphere. Similarly, a sphere with an axial surface-current-density proportional to the cosine of the latitude will also have a uniform interior magnetic field.* This type of surface-current-density can be approximated by winding a coil on the outside of a sphere with the density of windings just equal to the cosine of the latitude. The magnetic field intensity within a sphere of radius R with current i through an axial coil of N turns is

$$H = \frac{\mu_o Ni}{3R} \tag{1}$$

If the axis of the sphere is parallel to the earth's magnetic field, it is possible to cancel out the magnetic field within the sphere by adjusting the current through the coil.

A spinning dipole within the sphere will cause a voltage to be induced in the outside spherical coil. In order to calculate the magnitude of the voltage induced by the spinning dipole, it is necessary to first find the voltage induced in a circular loop of wire, of area A, at an angle θ to the axis of the measuring coils, due to changes in the current measuring coil.

The magnetic flux through the loop is

$$\Phi = \int \bar{N} \cdot \bar{H} dA = A\,H \cos\theta$$

Hence, the induced voltage inside the loop is

$$E = -\frac{d\Phi}{dt} = \frac{A \cos\theta\, N\mu_o}{3R}$$

Thus, the mutual inductance between the sphere windings and the loop is

$$L_{12} = L_{21} = \frac{A \cos\theta\, N\mu_o}{3R}$$

*"Static and Dynamic Electricity," William Smythe, Second Edition, McGraw-Hill Book Company p. 274.

The voltage induced in the outside windings of the sphere by a spinning dipole moment of strength IA can be determined from this equation. If a current is considered to flow in the windings of I and if the loop is spinning about an axis perpendicular to the axis of the sphere at a rate ω, the induced voltage in the outside coil will be:

$$E_d = \frac{dL_{21}\, I}{dt} = \frac{I\, AN\omega\mu_o}{3R} \sin \omega t \qquad (2)$$

This expression shows that the measured voltage has frequency ω and that the magnitude is proportional to both the dipole strength (IA) and the speed of rotation. The voltage induced by a satellite spun at a known speed ω inside the spherical coil will determine the satellite's dipole moment.

A second voltage will be induced in the measuring coils due to eddy currents in the satellite and metal yoke which spins the satellite. These eddy currents are caused by the voltages which the earth's magnetic fields induce in closed metal loops. Most of the current loops originate in the metal yoke which holds the satellite in the test apparatus. Although the induced eddy currents caused by the satellite's spin in space will also create a torque, the effect of this torque is small compared to the non-induced dipole torque. The second-harmonic or eddy-current induced voltage, will be superimposed upon the voltage caused by the dipole moment and, hence, will constitute noise. In order to estimate the magnitude of the second harmonic, it is necessary to consider the satellite and yoke as being replaced by a single metal ring of area A and electrical resistance R_e.

The eddy current induced in the ring is

$$I' = \frac{1}{R_e} \frac{d\Phi}{dt} = \frac{H_e A\omega}{R_e} \cos \omega t \cos \varphi$$

where: H_e earth's magnetic field

φ angle between the axis of rotation and the earth's magnetic field

The induced voltage due to eddy current is

$$E_n = \frac{dI'\, L_{21}}{dt} = \frac{\mu_o A^2 \omega^2 NH_e}{3RR_e} \cos \varphi \sin 2\omega t \qquad (3)$$

An interesting feature of this equation is that the frequency of induced voltage is twice that of the voltage induced by the dipole. Also, the magnitude is proportional to ω^2, which means that a better signal-to-noise level can be obtained by decreasing the speed of rotation. Evaluation of equation 3 for some typical values of the parameters shows that, due to the large holding yoke, the induced eddy current signal can be larger than the dipole signal. For this reason, it is necessary to perform the experiment in a field-free space. In order to approximate this condition a second coil of wires, identical to the first, is wound on the sphere, and the sphere's axis is aligned with the theoretical earth's field. By passing a current through the second outside coil, a field which is equal and opposite to the earth's field can be produced inside the sphere. This insures against the existence of a magentic field within the sphere and thus prevents eddy current signals from being induced in the measuring coils.